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# *Journal*

*of*

# THE ROYAL AERONAUTICAL SOCIETY

FEBRUARY 1958


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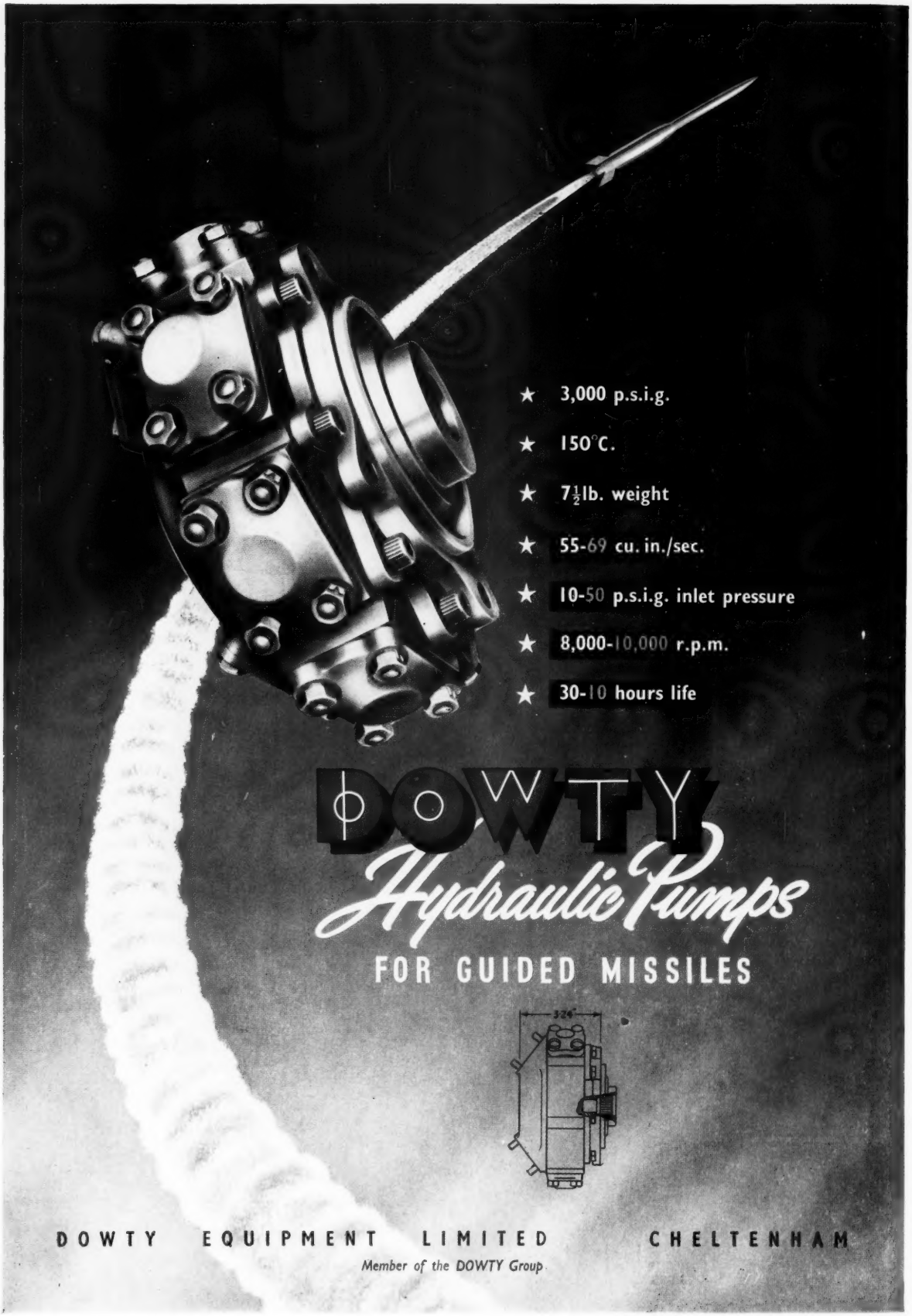
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*All communications for publication in, and advertisements for, the* JOURNAL *should be addressed to the Editor,* JOAN BRADBROOKE, *A.R.Ae.S.; all communications on general matters affecting the Society should be addressed to the Secretary,* A. M. BALLANTYNE, *T.D., B.Sc., Ph.D., Hon.F.C.A.I., A.F.I.A.S., F.R.Ae.S.*

*None of the papers or paragraphs must be taken as expressing the opinion of the Council of the Royal Aeronautical Society unless otherwise stated.*

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# NOTICES

### Presidential Address and Reception
### Wednesday 26th February 1958

Sir George R. Edwards will deliver his Presidential Address on Wednesday 26th February 1958 at Church House, Westminster, London, S.W.1 (entrance in Great Smith Street), at 6.0 p.m. Lord Hives will take the Chair and during the evening the Shell Petroleum Company's film "High Speed Flight—Part II" will be shown for the first time. The Address will be followed by a Reception at 4 Hamilton Place, London, W.1. An application form for tickets for the Reception has been sent to members.

### Nomination of Candidates for Council

The following is an extract from the By-Laws:—

"The Twenty-one ordinary members (of the Council) shall be nominated and elected from among the members of the Society. At the date of their election at least ten shall be Fellows, and one at least shall be in each of the following classes: Associate Fellow, Associate and Graduate.

"Of the ordinary members of the Council that number necessary to create seven vacancies shall retire annually. The retiring members shall be those with the longest service since their election but they shall be eligible for re-election.

"Nominations of candidates for election to the Council must be received by the Secretary not later than 10th April in each year and shall include statements in writing by the candidates that they are willing to serve. The nomination forms shall be signed by one proposer and two seconders, all of whom shall be Voters."

Nomination forms may be obtained from the Secretary.

### Radio Aids to Aeronautical and Marine Navigation

A convention on Radio Aids to Aeronautical and Marine Navigation is to be held on 27th and 28th March by the Radio and Telecommunication Section of the Institution of Electrical Engineers. Subjects to be covered include: Ground, Air and Shipborne Radar; Harbour and Airfield Approach Aids; Hyperbolic and Distance Bearing Navigational Aids; and Doppler and Inertia Navigation.

Members of the Society are invited to attend the sessions and those interested should notify the Secretary, The Institution of Electrical Engineers, Savoy Place, London, W.C.2, as quickly as possible so that they may receive particulars of registration fees and other details.

### International Analogy Computation Meeting

The Second International Analogy Computation Meeting, under the patronage of the "Association Internationale pour le Calcul Analogique," is to be held at Strasbourg from 1st to 9th September 1958, and not in June as previously announced. There will be an exhibition of analogy computation machines and equipment from 30th August until 10th September. The meeting will cover the various Analogy Computation Methods (differential analysers, rheoelectrical analogies, network analysers, simulators, special calculators, and so on) and their applications to science and industry. A new group devoted to electronic digital computation is also to be formed.

The Association's representative in the United Kingdom is Professor S. C. Redshaw, *Fellow*, but full particulars of the Strasbourg meeting should be obtained from Monsieur F. H. Raymond, Secretaire Delegue General de l'Association Internationale pour le Calcul Analogique, 138 Boulevard de Verdun, Courbevoie (Seine), France.

### New Year's Honours List

| | |
|---|---|
| *Knight Bachelor* | G. J. R. L. D'Erlanger (*Associate*) |
| *K.C.B.* | Air Vice-Marshal A. W. B. McDonald (*Associate Fellow*) |
| *C.B.* | Hayne Constant (*Fellow*)<br>J. E. Serby (*Fellow*) |
| *C.B.E.* | Captain V. A. M. Hunt (*Fellow*)<br>W. A. Summers (*Fellow*) |
| *O.B.E.* | L. R. E. Appleton (*Fellow*)<br>A. D. Balmain (*Associate Fellow*)<br>R. G. Fall (*Associate Fellow*)<br>I. B. Fleming (*Fellow*)<br>Captain R. P. Mollard (*Associate Fellow*)<br>E. J. Richards (*Fellow*)<br>J. L. Watkins (*Fellow*) |

### The Eleventh Louis Bleriot Lecture

The Eleventh Louis Bleriot Lecture will be delivered in Paris on Wednesday 5th March 1958. The Lecture, entitled "The Future of Aeronautical Research," will be given by Professor W. J. Duncan, C.B.E., D.Sc., F.R.S., F.R.Ae.S., at 5.0 p.m. at L'Aero Club de France, 6 rue Galilée, Paris XVI.

### Associate Fellowship Examinations—June 1958

Entries for the next Associate Fellowship Examination should be received from Candidates in the British Isles by 28th February 1958. The closing date for entries outside the United Kingdom was 31st December 1957.

### American Honour Awarded to a Member

Dr. Bo. K. O. Lundberg (*Fellow*) has been awarded Honorary Fellowship of the Institute of the Aeronautical Sciences. The Honour is the highest the Institute can bestow. Dr. Lundberg is Director of the Aeronautical Research Institute of Sweden.

### Acknowledgments

The Council wish to thank Vickers-Armstrongs (Aircraft) Ltd. for a set of blueprints of the Vickers Vimy Commercial aeroplane.

They also thank Geoffrey Dorman, Esq., *Associate*, for presenting a number of photographs and other historical material and C. Smith, Esq., *Associate Fellow*, for returning back numbers of the Journal.

### Royal Society of Arts Prizes

The following prizes are offered by the Royal Society of Arts in 1958:—

The Howard Prize of £50 for a treatise on some aspect of the subject of motive agents, and the Fothergill Prize of £20 for a descriptive essay or model embodying some new idea for the prevention or suppression of fire. The closing date for entries is the 31st July 1958. Full particulars and entry forms may be obtained from The Secretary, Royal Society of Arts, John Adam Street, Adelphi, London, W.C.2.

### The Use of Magnesium in Aircraft Structures

The Society of Aircraft Materials and Process Engineers (Los Angeles) is planning a symposium in April in conjunction with the Magnesium Association and the Dow Chemical Company on the use of magnesium in Aircraft Structures.

Further particulars of the meeting can be obtained from the Secretary, Mr. Sam Rutherford, at the Society of Aircraft Materials and Process Engineers, Los Angeles.

# DIARY

## LONDON

**13th February**

MAIN LECTURE.—The Application of the Theory of Stability in Structural Design. H. L. Cox. Institution of Mechanical Engineers, Birdcage Walk, S.W.1. 6 p.m.

18th February

SECTION LECTURE.—Aeroelastic Problems Associated with High Speeds and High Temperature. E. G. Broadbent. Library, 4 Hamilton Place, W.1. 7 p.m.

20th February

MAIN LECTURE AT HALTON BRANCH. THE FIRST TRENCHARD MEMORIAL LECTURE.—Marshal of the R.A.F. Sir Dermot Boyle. Burnett Gymnasium, R.A.F. Halton. 6.45 p.m.

26th February

PRESIDENTIAL ADDRESS. Sir George R. Edwards. Church House, Westminster, S.W.1. 6 p.m. Followed by reception (tickets required) at 4 Hamilton Place.

**5th March**

THE ELEVENTH LOUIS BLÉRIOT LECTURE. IN PARIS

The Future of Aeronautical Research. Professor W. J. Duncan. L'Aero Club de France, 6 rue Galilée, Paris XVI. 5 p.m.

13th March

MAIN LECTURE AT CHRISTCHURCH BRANCH.—Flight Tests of a Meteor Aeroplane fitted with Jet Deflection. P. F. Ashwood and D. Lean. Bournemouth Municipal College. 7.30 p.m.

27th March

THE THIRTEENTH BRITISH COMMONWEALTH LECTURE.—Why Airlines are Hard to Please. B. S. Shenstone. Institution of Civil Engineers, Great George Street, S.W.1. 6 p.m. (Tea at 5.30 p.m.)

### GUIDED FLIGHT SECTION

**25th February**

Guided Flight Trials. R. W. Boswell. **Londonderry House, 19 Park Lane. 6 p.m.**

**20th March**

Problems in the Development of a Guided Missile. J. Clemow. 4 Hamilton Place, W.1. 7 p.m.

### GRADUATES' AND STUDENTS' SECTION

**25th March**

Annual General Meeting followed by Film Show. 4 Hamilton Place, W.1. 7.30 p.m. (Light refreshments 7 p.m.)

## BRANCHES

**11th February**

**Boscombe Down**—Film Evening. Lecture Hall, A. and A.E.E. 5.45 p.m.

12th February

**Chester.**—Joint Meeting with Stanlow Branch of Institute of Petroleum. Aircraft Fuelling Problems. E. L. Bass. Lecture Theatre, Grosvenor Museum, Chester. 7.30 p.m.

13th February

**Cheltenham.**—The Microscope in Engineering. A. E. Bingham. St. Mary's College, Cheltenham. 7.30 p.m.

**Bristol.**—Production of Civil Aircraft. P. Imlach. Filton House, 6 p.m.

17th February

**Henlow.**—N. E. Rowe Medals—Junior Members Lecture Competition. Building 62, R.A.F. Technical College. Henlow. 7.30 p.m.

19th February

**Christchurch.**—Aircraft Carriers. A. J. Simms. Kings Arms Hotel. 7.30 p.m.

**Leicester.**—The Activities of the P.E.R.A. Organisation. R. Tilsley. Lecture Theatre, Loughborough College. 6.45 p.m.

**Southampton.**—Branch Prize Papers. Institute of Education, University of Southampton. 7 p.m.

20th February

**Bristol.**—Sale of Civil Aircraft. R. White-Smith. Filton House. 6 p.m.

**Halton.** — MAIN SOCIETY LECTURE. First Trenchard Memorial Lecture. Marshal of the R.A.F. Sir Dermot Boyle. Burnett Gymnasium, R.A.F. Halton. 6.45 p.m.

**Isle of Wight.**—Modern Methods of Air-Sea Rescue. Lt. Cdr. J. S. Sproule. Clubhouse, Saunders-Roe Sports and Social Club, E. Cowes. 6.30 p.m.

**Weybridge.**—Some Interesting Problems in Ship-Building Today. R. J. W. Rudkin. Apprentice Training School of Vickers-Armstrongs (Aircraft) Ltd., Weybridge. 6 p.m.

26th February

**Hatfield.**—The Fairey Delta 2. R. L. Lickley and L. P. Twiss. de Havilland Restaurant, Hatfield. 6.15 p.m.

**Preston.**—Problems Associated with Non-Metallic Materials in Aircraft. N. J. L. Megson and E. W. Russell. R.A.F. Association, Preston. 7.30 p.m.

27th February

**Bristol.**—Servicing of Civil Aircraft. I. J. Gregory. Filton House. 6 p.m.

**Cambridge.**—Some Aspects of Aircraft Production. H. Povey. No. 1 Lecture Theatre, Cambridge University Engineering Laboratories. 8.15 p.m.

**Yeovil.**—THIRD HENSON AND STRINGFELLOW MEMORIAL LECTURE. S. G. Hooker. Park School, Park Road, Yeovil. 7.30 p.m.

28th February

**Birmingham.**—Branch President's Night. F. F. Crocombe. Engineering Centre, Birmingham. 7.30 p.m. **N.B. New Date**

**3rd March**

**Cheltenham.**—Investigation of Flutter and Vibration on Aircraft. W. G. Molyneau (by invitation of The Society of Instrument Technology). Rotunda, Cheltenham. 7.30 p.m.

**Derby.**—A Propulsion Symposium. Rolls-Royce Welfare Hall, Nightingale Road, Derby. 3 p.m.-5.30 p.m.; 6 p.m.-8.30 p.m.

**Henlow.**—Nuclear Propulsion for Aircraft. Professor A. D. Baxter. Building 62, R.A.F. Technical College. Henlow. 7.30 p.m.

4th March

**London Airport.**—2nd Annual Dinner.

**Luton.**—Ultra Sonic Testing of Light Alloys. J. Crowther. Napier Senior Staff Canteen, Luton Airport. 6.15 p.m.

5th March

**Brough.**—Human Limitations of High Speed Flight. Wing Cdr. P. D. G. V. Whittingham. Lecture Hall, Electricity Offices, Ferensway, Hull. 7.30 p.m.

**Reading.**—Industrial Efficiency. H. S. Wilkinson. Western Manufacturing (Reading) Ltd. 6 p.m.

6th March

**Isle of Wight.**—Transient High Acceleration and the Human Body. Wg. Cdr. F. Latham. Clubhouse, Saunders-Roe Sports and Social Club, E. Cowes. 6.30 p.m.

10th March

**Halton.**—Supersonic Aircraft and Missiles. P. J. Duncton. Branch Hut, R.A.F. Halton. 6.45 p.m.

11th March

**Boscombe Down.**—Navigation and the Air Traffic Control Problem. D. O. Fraser. Lecture Hall, A. and A.E.E. 5.45 p.m.

**Glasgow.**—Next Generation of Civil Aircraft. G. H. Lee. Room 24, College of Science and Technology, Duke Street, Glasgow. 7.15 p.m.

12th March

**Chester.**—The Investigation of Aircraft Accidents. E. L. Ripley. Lecture Theatre, Grosvenor Museum, Chester. 7.30 p.m.

**Gloucester.**—Problems of the Large Helicopter. Dr. G. S. Hislop. The Wheatstone Hall, Brunswick Road, Gloucester. 7.30 p.m.

**Southampton.**—Future Developments with Satellite Rockets. L. R. Shepherd. Institute of Education, University of Southampton. 7 p.m.

**Swindon.**—Annual General Meeting and "Viscount" films. The College, Victoria Road, Swindon. 7 p.m.

**Weybridge.**—Aircraft Production; Present and Future. Joint meeting with the Institute of Mechanical Engineers. S. P. Woodley. Apprentice Training School, Weybridge. 6 p.m.

13th March

**Christchurch.**—MAIN SOCIETY LECTURE. Flight Tests of a Meteor Aeroplane fitted with Jet Deflection. P. F. Ashwood and D. Lean. Bournemouth Municipal College. 7.30 p.m.

17th March

**Bristol.**—BARNWELL MEMORIAL LECTURE. The Economy of Structures. Sir Alfred Pugsley. Bristol University Engineering Lecture Theatre. 7 p.m.

**Henlow.**—Film Evening. Building 62, R.A.F. Technical College. 7.30 p.m.

19th March

**Leicester.**—Aircraft Photography 1913-1957. John Yoxall. Lecture Theatre, Loughborough College. 6.45 p.m.

20th March

**Cambridge.**—The Pioneers. N. J. Capper. No. 1 Lecture Theatre, Cambridge University Engineering Laboratories. 8.15 p.m.

**Isle of Wight.**—The Development of Power Plants for Supersonic Flight. Dr. E. S. Moult, J. Edwards and W. Neat. Clubhouse, Saunders-Roe Sports and Social Club, E. Cowes. 6.30 p.m.

**Yeovil.**—A Review of Private Flying. A. Elsmore. Park School, Park Road. 7.30 p.m.

21st March

**Birmingham.**—A Talk on Astronomy. M. W. Ovenden. Engineering Centre, Birmingham. 7.30 p.m.

25th March

**Luton.**—Annual General Meeting and Presidential Address. Napier Senior Staff Canteen, Luton Airport. 6.15 p.m.

26th March

**Hatfield.**—Civil Airworthiness and the Air Registration Board. R. E. Hardingham. de Havilland Restaurant. 6.15 p.m.

**Preston.**—American Production Methods. A. W. Turner. Queen's Hotel, Lytham. 7.30 p.m.

27th March

**Bristol.**—Annual General Meeting and Film Show. Filton House. 6 p.m.

**Glasgow.**—Application of Electronic Control Techniques to Aircraft. Member of Staff of Ferranti Ltd., Edinburgh. Room 24, College of Science and Technology, Duke Street, Glasgow. 7.15 p.m.

## AGARD Structures and Materials Panel and Other Meetings

The Seventh Meeting of the Structures and Materials Panel of the Advisory Group for Aeronautical Research and Development (N.A.T.O.) will take place in Rome from 24th March to 2nd April 1958.

The Meeting includes a three-day Symposium to an arranged programme on Non-Metallic Materials, the annual one-day meeting on Flutter and one-day discussion meetings on The Uses of Photoelasticity and on General Instability Problems.

Further details of the programme may be obtained from the United Kingdom Panel Members, namely, Mr. A. J. Barrett (Royal Aeronautical Society), Mr. F. A. Kerry (Saunders-Roe Ltd., representing the S.B.A.C.), and Mr. M. O. W. Wolfe (Structures Department, Royal Aircraft Establishment, Farnborough).

The Combustion and Propulsion Panel will hold its Third Colloquium in Rome, also, from 17th to 21st March. This meeting will be largely devoted to consideration of current propulsion problems.

The Wind Tunnel and Model Testing Panel will sponsor a meeting in Freiburg, Breisgau, Germany, from 22nd to 25th April which will commemorate the centenary of Carl Cranz.

## Annual Subscriptions

Members are reminded that their annual subscriptions became due on 1st January 1958. The rates are:—

| | Home | | | Abroad | | |
|---|---|---|---|---|---|---|
| | £ | s. | d. | £ | s. | d. |
| Fellows | 5 | 5 | 0 | 4 | 4 | 0 |
| Associate Fellows | 4 | 4 | 0 | 3 | 3 | 0 |
| *Associates | 3 | 3 | 0 | 3 | 3 | 0 |
| Graduates (aged under 26) | 2 | 2 | 0 | 2 | 2 | 0 |
| Graduates (aged 26 and over) | 2 | 12 | 6 | 2 | 12 | 6 |
| Students (aged under 21) | 1 | 1 | 0 | 1 | 1 | 0 |
| Students (aged 21 and over) | 1 | 11 | 6 | 1 | 11 | 6 |

* Any Associate elected before 1st October 1947 may, if he wishes, elect not to receive the Journal, and in this case his subscription will be reduced by £1 1s. 0d. to £2 2s. 0d.

It will avoid delay and confusion if members, when sending remittances for subscriptions, will state their names clearly and give their addresses and grades of membership. Remittances should be made payable to the Royal Aeronautical Society.

## Elections

The following is a list of new members and transfers of membership:—

*Associate Fellows*

Ray Miller Brown (*from Associate*)
Jagannath Punnulal Chawla
Arthur Horace Sidney Fernando (*from Graduate*)
Peter Field (*from Graduate*)
Roy Hawkins (*from Graduate*)
Frank Housley
David Johnson (*from Associate*)
Olle Ljungstrom
Vettakkorumakankav Sivarama Narayanan
James Michael O'Connell
B. N. Vaman Rao
Geoffrey Walter Webber (*from Graduate*)
Cyril Edgar Luke Wray (*from Associate*)

*Associates*

Arthur Ashton Fawcett
Harold Edward John Godden
Edwin James Harris
Paul Peter Heller (*from Companion*)

*Graduates*

John Churchill (*ex Student*)
Alfred Harold Flint
John David Gabbott (*from Student*)
Douglas Henry Jagger (*from Student*)
Vernon Marples (*from Student*)
John Michael Parkinson

*Students*

Colin Joseph Aldridge
K. T'Chandy Alexander
Terence Edward Base
John Richard Darbyshire
John Edwin Dibley
Bryan John Evans
Terence Edward Halsey
Barry Douglas Hobson
Martin Holmes
Roger Hurd
Peter William Hyde
James John Johnston
Michael Weston Jones
George William Kitching
Jai Gopal Kohli
Malcolm Fletcher Moulton
Brian Edward Nelson
Allen Parker
David Shields
Donald Rowland Smith
James George Fingland Smith
Anthony Roger Chennel Tatham
Peter Ronald White
Donald Ernest Wright

*Companion*

Eric Douglas Margetts

## News of Members

Wing Commander A. D. Balmain (*Associate Fellow*), formerly at R.A.F., Oldenberg, has been appointed Senior Technical Officer at the R.A.F. Station, Gaydon.

P. J. Bell (*Graduate*), formerly a member of the personal staff of the General Manager (G.W.) Fairey Aviation Co. Ltd., has been appointed Project Engineer with the firm's associated company, Atomic Power Constructions Ltd.

H. Dudley Collard (*Graduate*), formerly of Avro Aircraft Ltd., Ontario, is now an Associate Engineer with the Boeing Airplane Co., Transport Division, Renton, Washington.

A. Coull (*Graduate*), formerly of English Electric Co. Ltd., has been appointed to a Lectureship in Engineering at the University of Aberdeen.

J. R. Cownie (*Graduate*), formerly Technical Public Relations Officer, Handley Page Ltd., has joined the editorial staff of "The Aeroplane."

G. A. Cropper (*Graduate*), formerly with the Bristol Aeroplane Co., Filton, is now an Aerodynamicist with de Havilland Aircraft (Canada) Ltd., Ontario.

D. Dewar (*Associate Fellow*), of the Dunlop Rubber Co. Ltd., has been appointed Chief Designer of the Company.

Group Captain W. D. Disbrey (*Associate Fellow*) has been appointed Director of R.A.F. Bomber Aircraft Research and Development, Ministry of Supply, with acting rank of Air Commodore.

Squadron Leader N. F. Duke (*Associate*) is now associated in an advisory capacity with Field Aircraft Services Ltd. and Fieldair Ltd., mainly in connection with Ferrying Operations and Service Liaison.

P. J. DUNCTON (*Associate Fellow*), formerly Assistant Chief Technician of the Fairey Aviation Co. Ltd., has been appointed Assistant Chief Engineer (Atomics) to the company.

RONALD F. FERGUSON (*Graduate*) has been appointed Development Engineer at Vickers-Armstrongs (Aircraft) Ltd., Swindon.

Flight Lieut. G. B. FOORT (*Associate*), formerly D.A.Mech., Ministry of Supply, has been appointed to Tech.(Elect.) Duties at R.A.F., Acklington.

S. W. G. FOSTER (*Associate Fellow*) has relinquished his position as Technical Director of the Broadway Aircraft Engineering Co. Ltd. in order to devote the whole of his time to his Design Consultancy Practice.

Cdr. R. F. FREITAG (*Associate Fellow*), formerly at the Chief of Naval Operations Office, Washington, has now been posted to the U.S. Naval Air Missile Test Center, Point Mugu, California.

T. F. R. GEORGE (*Associate Fellow*), Assistant Chief Technician, Fairey Aviation Co. Ltd., has been appointed Head of Physics and Engineering Dept., Atomic Power Constructions Ltd., Research and Development Laboratory, the Fairey Company's associated firm.

V. GIRVIN (*Student*), formerly a Stressman at Electro-Hydraulics Ltd., is now Assistant Engineer in the Stress Office of English Electric Co. Ltd., Warton.

C. GOLDBERG (*Associate Fellow*), formerly at Saben Hart and Partners, is taking up a senior appointment on the Technical Staff of Dowty Equipment Ltd., Cheltenham.

T. C. GREAVES (*Associate Fellow*), formerly of Bristol Aeroplane Co. Ltd., is now a Senior Engineer at the English Electric Co. Ltd., Atomic Power Projects, Whetstone, Leics.

A. GRIFFITHS (*Associate Fellow*), formerly at the Constantine Technical College, has been appointed Lecturer in charge of Mechanical Engineering at the West Essex College of Further Education, Harlow.

E. HASSELL (*Graduate*), formerly at Gloster Aircraft Co. Ltd., is now employed as an Engineering Aerodynamicist at Avro Aircraft Ltd., Malton, Ontario.

W. G. HEARLE (*Associate Fellow*), formerly of de Havilland Aircraft Ltd., Hatfield, has been appointed Chief Estimator at the Chester Branch of the Company.

E. H. HOLDER (*Associate Fellow*), formerly Manager (W) Royal Ordnance Factory, has been appointed Training and Education Officer, M.O.S., London.

R. G. KNIGHT (*Associate Fellow*), formerly an Aerodynamicist, Fairey Aviation Co. Ltd., has been appointed Senior Research Assistant with the firm's associated company, Atomic Power Constructions Ltd., Research and Development Laboratory.

E. LEMBERG (*Associate Fellow*), formerly at Bristol Aircraft Ltd., is now a Stress Engineer at Avro Aircraft Ltd., Malton, Ontario.

W. B. LEWIS (*Associate Fellow*), formerly at the R.A.E., is now Assistant Director, Directorate of Civil Aircraft Research and Development, M.O.S.

J. R. LINGE (*Associate Fellow*), formerly Technical Assistant (Structs Test) Fairey Aviation Co. Ltd., has been appointed Group Leader with the firm's associated company, Atomic Power Constructions Ltd., Research and Development Laboratory.

I. S. MANT (*Associate Fellow*), formerly of Vickers-Armstrongs (Aircraft) Ltd., is now with the Research Department of Smiths Aircraft Instruments Ltd., Cheltenham.

K. J. MCKAY (*Associate Fellow*), formerly of the Ordnance Board, Charles House, is now Assistant Director Aircraft Armament, M.O.S., St. Giles Court.

J. L. MILNE (*Associate Fellow*), formerly at Short Bros. and Harland Ltd., is now with Canadair Ltd., Montreal, as Engineer "A" in the Preliminary Design Section.

H. A. MONEY (*Graduate*), formerly with Saunders-Roe Ltd., is now a Stressman with Folland Aircraft Ltd.

T. G. MORTON (*Graduate*) has now resumed his position of Technical Assistant in the Electro-Dynamic Department of the de Havilland Engine Co. Ltd., Hatfield, after completing his National Service.

Squadron Leader D. NIXON (*Associate*) has retired from the R.A.F. and has taken a position with the Gloster Aircraft Co. Ltd. at Moreton Valance, Glos.

W. PITSON (*Associate*), formerly of Bristol Aeroplane Co., has been appointed Quality Manager of Bristol Aero Engine Co. Ltd.

A. J. PRATT (*Associate*), formerly Inspector of West African Airways Ltd., has now taken an appointment as a Surveyor with the Air Registration Board.

G. REYNARD (*Associate Fellow*) is now in charge of the Automotive Section of Technical Services Branch of The British Petroleum Co. Ltd. He was formerly an assistant in the Aviation Section of the same Branch.

G. E. ROSE (*Associate*), formerly of Johnson and Fletcher, has now joined the Engineering Staff of Messrs. Stewarts and Lloyds of Rhodesia Ltd.

L. SMITH (*Associate Fellow*) has left the Medway College of Technology to take up a Lecturing appointment at the Technical College, Bulawayo, Southern Rhodesia.

B. SOLOWIJ (*Student*) has left de Havilland Aircraft (Pty) Ltd., Australia, and is now an Experimental Officer in the Weapons Research Establishment, in the Department of Supply, Salisbury, Australia.

E. LLOYD THOMAS (*Associate Fellow*) has resigned from Short Bros. and Harland Ltd., Belfast, and is now Chief Development Engineer with R. B. Pullin and Co. Ltd.

J. TINKLER (*Associate Fellow*), formerly with Saunders-Roe Ltd., has been appointed Lecturer in Aeronautical Engineering at Bristol University.

P. H. TOWNSHEND (*Associate Fellow*), formerly of de Havilland Aircraft Co. Ltd., is now Engineer "A" in the Research and Development Department, Canadair Ltd., Canada.

D. F. TOOLEY (*Associate*), formerly of Field Aircraft Services, has been appointed Head of Planning Provisioning (Repairs) with Rotax Ltd., Hemel Hempstead.

Group Captain E. E. VIELLE (*Associate Fellow*), recently retired from the R.A.F., has taken up the appointment of Chairman of Vielle-Shipman Ltd., Park Lane.

F. R. WARD (*Associate Fellow*), of Fairey Aviation Co. Ltd., has been appointed Group Leader with the firm's associated company, Atomic Power Constructions Ltd., Research and Development Laboratory.

L. C. E. WASPE (*Associate Fellow*), formerly of Rolls-Royce Ltd., has taken an appointment as Technical Manager with Gydar Ltd., Weston-super-Mare.

E. W. C. WILKINS (*Fellow*) has been made a member of the International Meteoritic Society and a Corporate Member of the Société Française de Métallurgie.

J. R. WILLIAMS (*Associate*), formerly of de Havilland Aircraft Co. Ltd., is now an Engineer in the Design Division of Canadair Ltd., Montreal.

### SYMPOSIUM ON RAREFIED GAS DYNAMICS AND AEROTHERMODYNAMICS

This Symposium will be held in Nice from the 2nd-5th July 1958. The Faculty of Science of the University of Paris has organised the Symposium in connection with the present position regarding the Research on Rarefied Gas Dynamics and Aerothermodynamics.

Further particulars may be obtained from the Director, Dr. F. M. Devienne, Laboratorie Méditerranéen De Recherches Thermodynamiques, 2 Avenue Villebois-Mareuil, Nice.

*The Journal of the Royal Aeronautical Society*

WITH WHICH IS INCORPORATED THE INSTITUTION OF AERONAUTICAL ENGINEERS

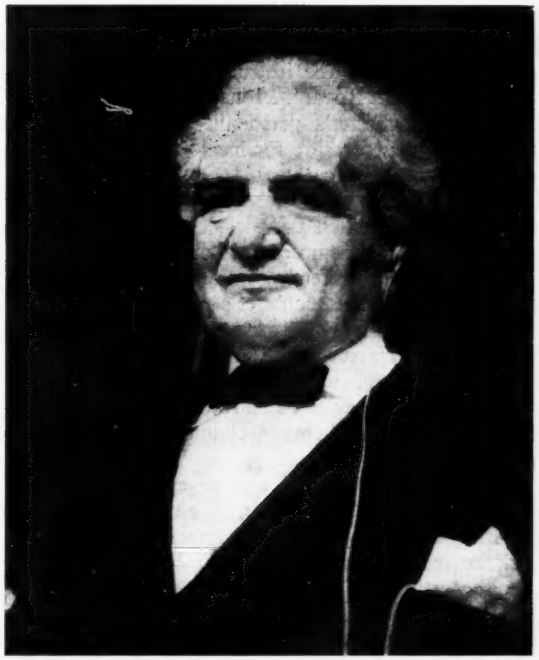
VOLUME 62 FEBRUARY 1958 NUMBER 566

# *The First Lanchester Memorial Lecture*

THE First Lanchester Memorial Lecture was given by Dr. Theodore von Kármán, Hon.F.I.A.S., Hon.F.R.Ae.S, at the Institution of Mechanical Engineers, Birdcage Walk, on 9th May 1957, before a large and distinguished audience which included Mrs. F. W. Lanchester and Mr. George Lanchester. Sir George Edwards, C.B.E., B.Sc., F.R.Ae.S., President of the Society, presided.

Sir George Edwards: This is the First Lanchester Memorial Lecture and it is also the first time that I perform in this capacity. I am very honoured at the fact that my first public appearance, so to speak, is one in which we pay tribute to a great Englishman. Aviation is an international thing and that this Society is international in its approach and attitude is, I think, best shown by the named lectures which are given at the centre of the Main Society as distinct from the Branches. Those named lectures so far have been the Wilbur Wright, honouring a great American; the Louis Bleriot, honouring a Frenchman and, of course, the Commonwealth Lecture.

I believe that it is right and fitting that we should on this occasion honour a great Englishman. The international touch is continued by the fact that we have the great fortune and pleasure in that Dr. von Kármán is giving the lecture tonight and, although I could say that he needs no introduction and his great name is such that everybody knows him, I think it is right that I should enumerate some of the features of this man's life. I don't think he will be cross with me if I tell you that he was born in Budapest in 1881, on 11th May, which, astounding though it might be, if my arithmetic is right, makes him 76 years young on Saturday and to see a man who has spent all his life in aviation looking like that at 76 tears one or two of us up. Budapest is a name that in the past few months has come very much to our minds and to our notice and I think maybe we do not have to look very far to see the roots which lead to the courage and the pioneering spirit and the independence of outlook which this man has always shown. He was born, as I say, in 1881 and he was publishing papers over 50 years ago. He graduated at the Royal Technical University at Budapest, moved across to Göttingen and studied there and then subsequently again at the University of Aachen. He moved to the United States in 1926 and as most of us know became established there in the Guggenheim Laboratory where he built up a great school and a great reputation for forging together theory and experiment in what was virtually a new type of engineering science. He has contributed papers in a great breadth of aeronautical subjects, too great for me to list. In 1944, General Arnold asked Dr. von Kármán to set up an Advisory Group for the United States Air Force and he became Chairman of the Scientific Advisory Board to the Chief of Staff. I know that under his Chairmanship that Board was responsible for recommending to the United States Air Force and their development groups a great many of the things that have put American aviation in the position which it now is. He was responsible for the establishment of, and was the first Chairman of, the Advisory Group on Aeronautical Research and Development within NATO, known to us as AGARD, and still is Chairman.



Dr. Theodore von Kármán, Hon.F.I.A.S., Hon.F.R.Ae.S. A photograph taken at the Lecture by Mr. P. S. Barna, A.F.R.Ae.S.

I have studied the works of reference on this man. One can almost devote a lecture to him as well as he devoting a lecture to Dr. Lanchester. But I think I have said sufficient and I am sure that we all extend a most warm welcome to you, sir, for coming to us on this occasion which we believe to be so important, and which you believe to be important too.

# Lanchester's Contributions to the Theory of Flight and Operational Research

by

THEODORE VON KÁRMÁN, Hon.F.I.A.S., Hon.F.R.Ae.S.

I CONSIDER it a great honour to be invited to deliver the First Lanchester Memorial Lecture, which, according to the instruction received from the Secretary of the Royal Aeronautical Society, should deal with Lanchester's personality and achievements. It is also a great pleasure for me to recall my personal contacts with this great man. On the other hand, I am not quite sure whether the decision of the Council, choosing me as the first Lanchester Lecturer, was really a wise one, since I am not in the best position to present a full biographical sketch of Lanchester. Somebody more familiar with Lanchester's automobile work, as, for example, Sir Harry Ricardo, would be more able to present a full picture of Lanchester's achievements. I must, therefore, restrict myself essentially to the treatment of Lanchester's contributions to the sciences of Aerodynamics, Flight Mechanics and Operational Analysis.

At this point I would like to express my sincere gratitude to several persons who either helped me in gathering data, reading and interpreting Lanchester's work, or reading the manuscript and contributing valuable suggestions. I would like to mention especially Mr. Harold Luskin of Douglas Aircraft and Mr. Karl Bergey of North American Aviation, both of whom collaborated with me in the preparation of the manuscript. Then I have to mention with gratitude my old friends and scholars of aviation history, J. Laurence Pritchard and A. R. Low.

Lanchester was born in 1863 and died in 1946. I assume that he was a very successful engineer, inventor, designer, and manager. His engineering contributions were generally in the field of automotive engineering. At the age of twenty-one he became Assistant Works Manager at Messrs. T. B. Barker's of Saltley, Birmingham, and engaged himself in efforts towards the development of the gas engine. He invented a starter during this period. The construction of the first Lanchester motor car began in 1894, and in 1899 the Lanchester Motor Company was formed with Lanchester as Chief Engineer and General Manager. From 1904 to 1914 he acted as consulting engineer to this company; from 1909 to 1929 he was consulting engineer and technical advisor to the Daimler Company and the Birmingham Small Arms Company; and from 1928 to 1930 he was consulting engineer to William Beardmore's on Diesel engines.

I would like simply to list the inventions and improvements due to him, as I found them in the literature: an engine starter (already mentioned), a surface carburettor, an epicyclic speed-change gear, a pre-selector gear, and apparatus for measuring worm gear efficiency, a wire wheel, an electric ignition device, a direct-driven top gear and normal transmission on a rear axle, a crankshaft damper, an harmonic balancer, a process for the manufacture of piston rings, a pendulum accelerometer for measuring and recording traction and braking effort, a gyroscopic true plumb and turn indicator (widely used by the Royal Air Force in the First World War), and an acoustic loud speaker and "diffractaphone aperture" for use both in concert halls and in the open air.

F. W. Lanchester, Honorary Fellow.

I am not able to say how many of these inventions are used today in the automotive and other industries. However, according to the testimony of contemporary writers and students of engineering history, they represented essential improvements in their particular time period.

His study of mechanical flight began as a side line. In fact, Captain Pritchard relates that in 1936 Lanchester

wrote to him, in answer to a request for an outline of what he had done: "My serious study of the problem of mechanical flight began in the year 1891. Sir Dugald Clark"—with whom Lanchester worked in close association in the gas engine field—"told me that, though he believed in what I was doing, that I should do well to keep it to myself, otherwise I should get the reputation of being a visionary, if not a madman."

Whether or not he was a visionary depends on the interpretation of the word. If it means a man who has an extraordinary vision—a deep insight into the problems he considers—he certainly was a visionary. However, he was *not* a visionary in the sense that in following his imagination he would overlook realities, or difficulties in realisation. On the contrary, his predictions—for example, those concerning possible developments and performances of aircraft, as we shall see later—were rather too cautious and certainly not "visionary," as I am afraid certain predictions of some of our brethren engaged in Astronautics are today.

Nevertheless, the first presentations of his "scientific vision" of the flight problem were in fact rejected. In 1894 he gave a talk before the Birmingham Natural History and Philosophical Society. Unfortunately, I have not been able to find a text of his lecture. According to Captain Pritchard, every time Lanchester moved into a new house he made a big bonfire of manuscripts and letters; probably he destroyed the manuscript of the paper. Lanchester himself says that he revised this paper in which he expanded the "vortical" theory of sustentation during the next three years and in 1897 submitted it to the Royal Society of London. It seems that some member of this Society thought that the paper would better suit the publications of the Physical Society. Lanchester followed this advice, but the Physical Society rejected his paper in the same year. The paper essentially contained, as Lanchester said, the fundamental concepts of the circulation theory of lift (sustentation) and aerodynamic drag (resistance) as they were presented in the first volume, entitled "Aerodynamics," of the "Aerial Flight," which was published ten years later, in 1907.

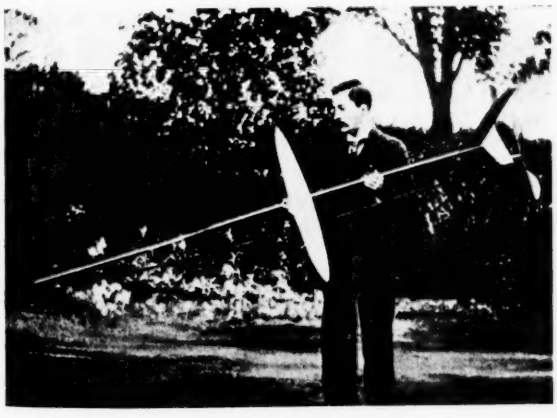
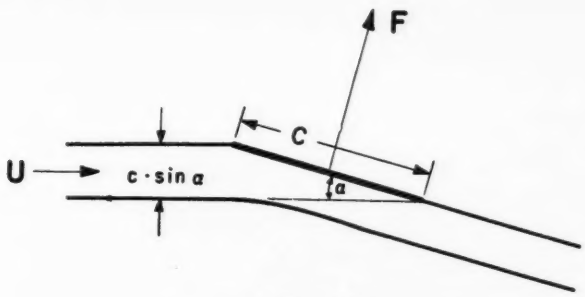


FIGURE 1. F. W. Lanchester with one of his model gliders (From Sutton, *Science of Flight*).




$$F = c\rho U^2 \sin^2 \alpha$$

FIGURE 2. Newton's theory.

In order to understand why it was so difficult for Lanchester to have his ideas concerning aerodynamic forces accepted, one must consider the state of theoretical and experimental knowledge in the last decades of the nineteenth and the first years of the twentieth centuries, before and immediately after the first powered flight was performed in 1903. The first theoretical computations of the forces exerted by the air on an inclined flat plate as a kind of idealised wing were based on the so-called Newton's theory, which is shown in diagram form in Fig. 2. Newton considered in general the impact of an idealised fluid on a solid surface. According to Giacomelli in Vol. I of Durand's *Aerodynamic Theory* (1934), John Robison (*A System of Mechanical Philosophy*, 1822), quoted by Pritchard in his paper The Dawn of Aerodynamics*, and others, Newton never considered air. According to A. R. Low he considered the air as a cloud of non-interfering particles. For this case he obtains a force which acts in the direction normal to the surface which is proportional to the density of the fluid, to the square of the relative velocity, to the surface area, and to the square of the sine of the inclination of the surface relative to the direction of the flow. In consequence of the assumption that the particles hit the surface without interfering with each other, Newton assumes that every surface element deflects the portion of the fluid stream which directly impinges on that element, and that the change of momentum in unit time due to this deflection determines the force.

It was found experimentally at a rather early date that the force acting on an inclined plate exposed to moving air, at least for small angles of inclination, is proportional to the sine of the angle (or approximately to the angle) rather than to the square of the sine. It was evident to the researchers, both the scientists and the flight enthusiasts, that if the sine square law were correct, the outlook for the feasibility of mechanical flight would be rather dim, since a favourable lift to drag ratio would require small angles of inclination, and small angles of inclination would mean enormous wing surface areas because of the extremely small value of the square of the angle. Another result of the

*Journal of the Royal Aeronautical Society*, March 1957.

FIGURE 3. Sir George Cayley.

mathematical theory which seemed to be contradictory to the facts was the so-called "d'Alembert's paradox." This great French mathematician and philosopher found that a solid body moving uniformly in a non-viscous fluid does not suffer any resistance.

In view of these theoretical results, which were obviously contradictory to observations, one cannot be surprised to read in the Fourteenth Annual Report of the Aeronautical Society of Great Britain (which, as I remember, was the predecessor of the Royal Aeronautical Society) the following statement: "Mathematics up to the present day has been quite useless to us in regard to flying." This was written in 1879.

On the other hand, Sir George Cayley, in the first decade of the nineteenth century, clearly defined the concept of lift and drag and announced the principle of the present-day aeroplane: sustentation of the weight by lift and compensation of the drag by a propulsion device actuated by an engine. Cayley also studied shapes leading to small drag. He proposed to learn from nature, and measured carefully the distribution of the cross-sectional area of the body of a trout along the length of the body. His proposed cross-sectional area distribution shows a striking similarity to the thickness distribution of some modern low-drag aerofoils. Fig. 4 illustrates this similarity.

Further, concerning the question of sustentation, two empirically obtained results, which were obviously important for any attempt to design aeroplanes, had no theoretical explanations: first, that the force on a flat wing is proportional to the angle of attack instead of being proportional to its square; and second, that a cambered plate, similar to the profiles investigated by Horatio Phillips, as illustrated in Fig. 5, and used by Otto Lilienthal in his glider experiments, shows a positive lift at zero angle of attack.

Lanchester's paper of 1894 gave an explanation for both these facts. The explanation for the linear proportionality with angle of attack is based on an assumption which he calls the assumption of constant sweep. The word "sweep" denotes the amount of fluid which is involved in the deflection caused by the profile. According to Newton's concept of the idealised fluid, the fluid volume deflected is proportional to the projected area of the plate on a plane perpendicular to the stream; in other words, it is proportional to the chord multiplied by the sine of the angle of inclination. According to Lanchester's concept, the volume of fluid involved in the process (the "sweep") is independent of the angle of attack; therefore, the force is proportional to the angle of deflection, which itself is equal, or proportional to, the angle of attack. Furthermore, he explains that the cambered surface causes an upwind in front of the leading edge and a downwind behind the trailing edge, as shown in Fig. 6. Since no accumulation of fluid downward or upward can take place, the result is a circulatory motion, which produces a higher pressure acting on the lower surface and a lower pressure on the upper surface of the wing.

It remains, nevertheless, a remarkable fact that Lanchester's paper in 1897 could be refused by the learned societies of his country. The idea of a force caused by circulation acting on a body moving through the air was not completely unknown. The effect of a spin or rotation on the trajectory of a projectile was studied experimentally by ballisticians. In a lecture delivered in 1746, Benjamin Robins, sometimes called the "father of gunnery," makes the following remarks: "I have only now to add, that, as I suspected, the consideration of the revolving

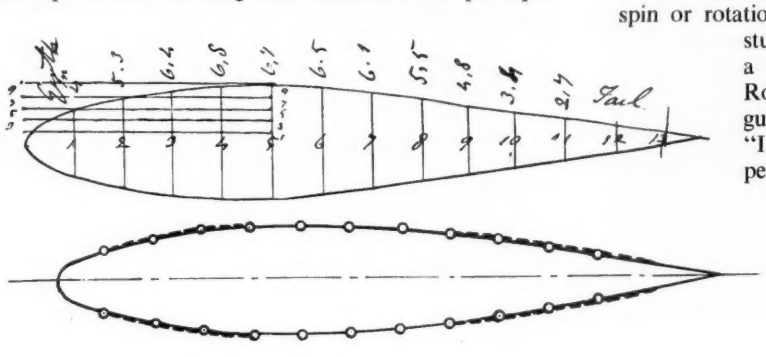



FIGURE 4. Cayley's sketch of the cross-section of the trout and its comparison with a modern low-drag aerofoil, (From *Aerodynamics*, von Kármán.)

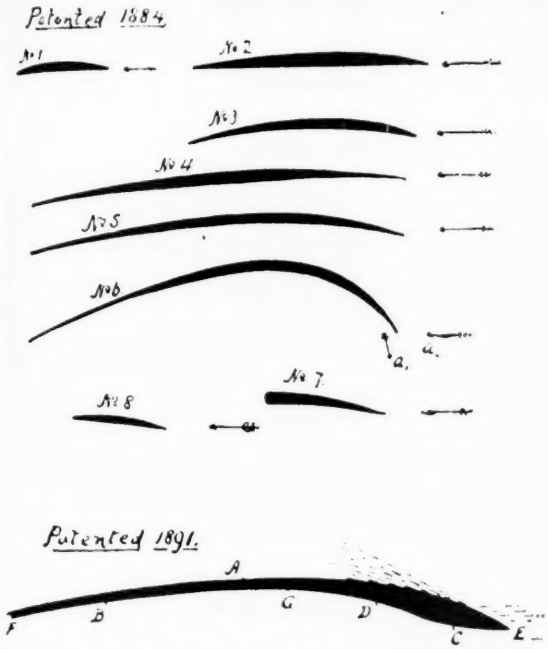



FIGURE 5. Wing sections studied by Horatio Phillips. (From *American Engineer and Railroad Journal*, 67 (1893), 135, and *Aerodynamics*, von Kármán.)

motion of the bullet, compounded with its progressive motion, might be considered as a subject of mathematical speculation, and that the reality of any deflecting force, thence arising, might perhaps be denied by some computists upon the principles hitherto received of the actions of fluids." The deflecting effect of the spin of the projectile became more well known about one hundred years later under the name of "Magnus effect." In 1877 Lord Rayleigh published in "The Messenger of Mathematics" (Vol. VII, May 1877-April 1878, pp. 14-16, Macmillan, 1878) a paper "On the Irregular Flight of a Tennis-ball," explaining the influence on the trajectory of the circulation of the air around the ball. In this paper the motion of the initially uniform and parallel flow of an inviscid fluid around an infinitely long circular cylinder is investigated and the result found that the drag, or the force in the direction of the stream, is zero according to d'Alembert's theorem, but that, when a circulatory flow or "circulation" is superposed, a force perpendicular to the stream acts on the cylinder, and this force is proportional to the superposed circulation. Captain Pritchard quotes, in his recently published highly interesting paper "The Dawn of Aerodynamics,*" the following page from a letter written by Newton and published in the "Philosophical Transactions" No. 80 (19th February 1671/72): —

"Then I began to suspect whether the rays, after their trajection through the prisme, did not move in curve lines, and according to their more or less curvity tend to diverse part of the wall. And it increased my suspicion, when I remembered that I had often seen a tennis ball, struck with an oblique racket, describe such a curve line. For, a circular as well as a progressive motion being communicated to it by the stroak, its parts on that side, where the motions conspire, must press and beat the contiguous air more violently than the other, and there excite a reluctancy and reaction of the air proportionately greater."

It would appear that persons familiar with the basic history of the question of circulatory motion should have been able to recognise the fundamental point in Lanchester's demonstration.

One reason for the lack of understanding of Lanchester's presentation might have been his particular liking for his own terminology, which was different from that commonly used. The term "sweep" was his own creation; the cyclic or circulatory motion called flow with circulation in the modern scientific literature he called "peripteral motion." He also talked of a "forced wave," although the analogy, of a flow round an aerofoil with wave motion, consists only of the fact that in both cases no energy is taken from the fluid or transferred to it. In his "Aerodynamics" (1907) he first reproduced the results he found and explained in 1897; then he subsequently gave "a revision of the theory on more orthodox hydrodynamic lines."

One may recall the amusing scene in Goethe's "Faust"; when Faust explains to Marguerite his theory of God and the Universe, she answers somewhat naively: —

> "All that is fine and good, to hear it so;
> Much the same way the preacher spoke
> Only with slightly different phrases."

In his Wilbur Wright Memorial Lecture given before the Society in May 1926, Lanchester makes the remark: "At the time of my investigation in 1892, I had very little acquaintance with classic hydrodynamic theory, or should have immediately recognised the form of fluid motion in question as a cyclic system superimposed on a motion of translation, which identity was only recognised some three or four years later."

In the ten years between 1897 and 1907, between the date of the rejection of Lanchester's paper and the publication of his book, two mathematicians—Kutta, a German, and Joukowski, a Russian—completed the theory as far as the aeroplane wing of infinite span is

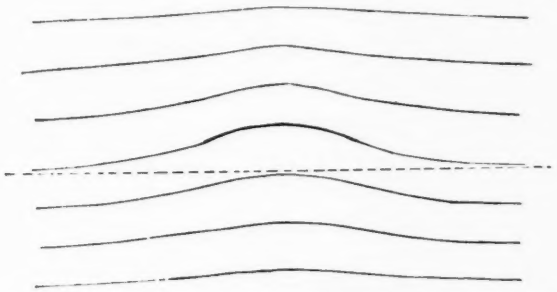


FIGURE 6. Lanchester's illustration of the effect of a cambered surface on the air flow. (*Aerodynamics*, Lanchester, Third Edition, 1911.)

*Journal of the Royal Aeronautical Society*, p. 152, March 1957.

concerned. Kutta was especially interested in Lilienthal's experiments and investigated the case of a cambered aerofoil, for example an arc, whereas Joukowski gave the general theorem for the magnitude of the lift force as connected with the magnitude of the circulation. Joukowski also investigated various aerofoil shapes with finite thickness. It is true that because of this coincidence of their efforts and Lanchester's delayed publications, Lanchester's clear priority for the circulation theory of sustentation was questioned; but it must be conceded that the mathematical theory gave more than he was able to deduce from his more or less qualitative arguments. On the other hand, he recognised the general mechanism of the lift and the induced drag also for the case of the aerofoil with finite span, for which the systematic mathematical theory needed approximately another decade of development.

Although we do not have the manuscript of Lanchester's 1897 paper, his patent taken out in the same year shows that he recognised the essential difference between the three-dimensional flow in the case of an aerofoil of finite span and the two-dimensional flow

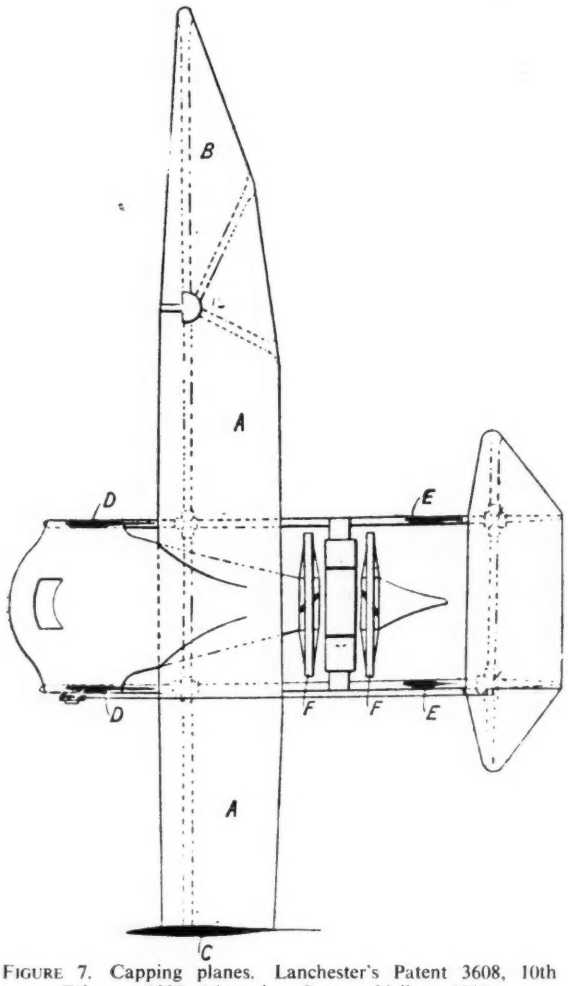



FIGURE 7. Capping planes. Lanchester's Patent 3608, 10th February 1897. (*Aeroplane Patents*, Neilson, 1910.)

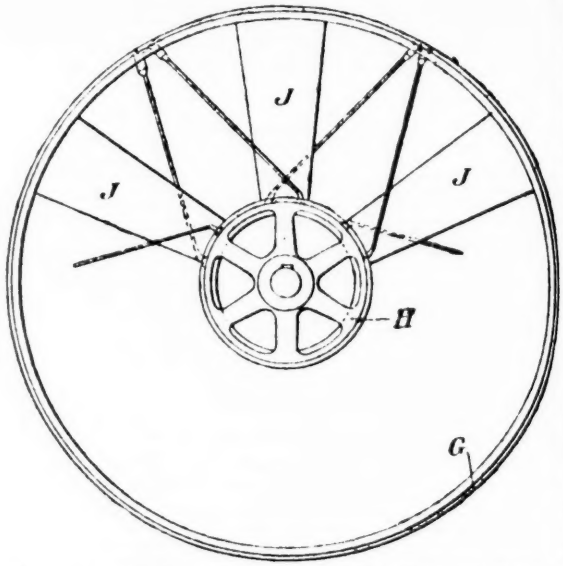



FIGURE 8. Lanchester's propeller. Patent 3608, 10th February 1897. (*Aeroplane Patents*, Neilson, 1910.)

in the cases of an aerofoil with infinite span and an aerofoil between two vertical parallel planes as far as energy loss is concerned. In his description of the patent, he suggests the arrangement of endplates, so-called "capping planes," located at the wing tips, as shown in Fig. 7. He evidently recognised the fact that to maintain sustentation in a non-viscid fluid by means of an aerofoil with infinite span does not require work. This was quite a revolutionary concept. As to the use of endplates, we remember that the German engineer Flettner later applied them to his rotorship and found that they materially increased the propulsive and steering forces produced by the rotors. It is also known that in present practice tip tanks and twin tails have a similar favourable influence on the induced drag. Lanchester extended his studies in wing theory to the design of propellers, one of which is shown in Fig. 8.

Although Lanchester never arrived at a complete theory of the finite wing, as did Prandtl several years later, the presentation of the problem in his book published in 1907 contains practically all the elements necessary for such a theory. He recognised that: —

1. In the case of a finite aspect ratio two vortex trunks are formed at the wing tips.
2. Each vortex trunk consists of a system composed of many individual vortices leaving the trailing edge of the wing along the span.
3. The kinetic energy contained in this vortex system must be renewed by expending work continuously during uniform flight.
4. The work required to maintain the same sustentation increases with decreasing aspect ratio.

The reason why he was unable to develop a quantitative theory of the lift distribution becomes clear if the representation of the vortex system in Lanchester's book, shown in Figs. 9-12, is compared with the schematic representation of Prandtl's concept in Fig. 13.

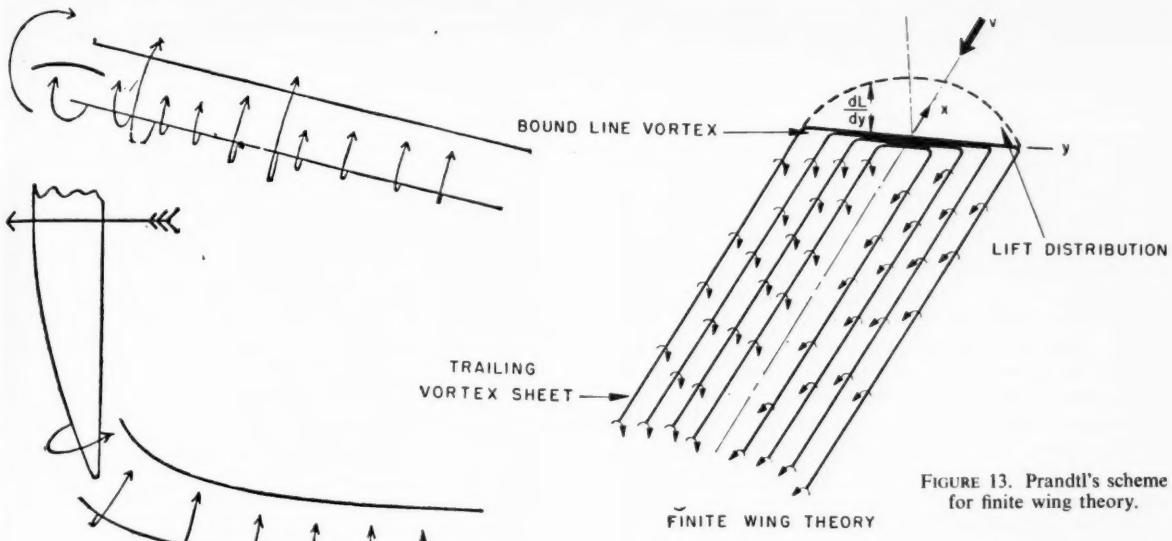



FIGURE 13. Prandtl's scheme for finite wing theory.

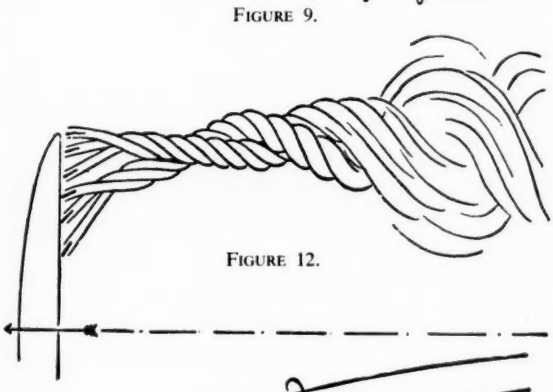


FIGURE 9.



FIGURE 12.

It appears that to compute Lanchester's vortex system would be extremely difficult even with all the means now available to mathematicians in the field of fluid mechanics. It must also be noticed that Prandtl's concept was simplified by the assumption that the velocities induced by the circulation were small in comparison with the flight velocity. This assumption allows the consideration of the trailing vortices as straight lines in the flight direction, whereas Lanchester constructs a resultant direction for the trailing vortices by the composition of two circulations: one around an axis perpendicular to the flight direction and one around an axis parallel to the flight direction. Furthermore, Lanchester's writing does not point out that the intensities of the trailing vortices are determined by the span-wise variation of the circulation along the wing.

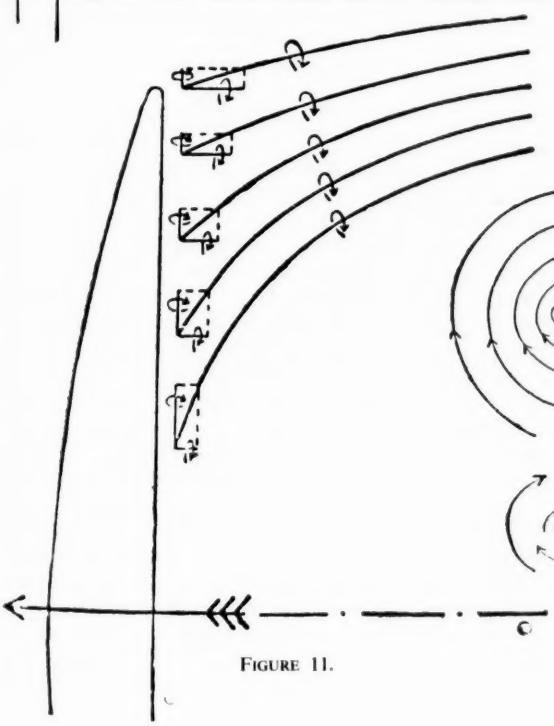


FIGURE 11.

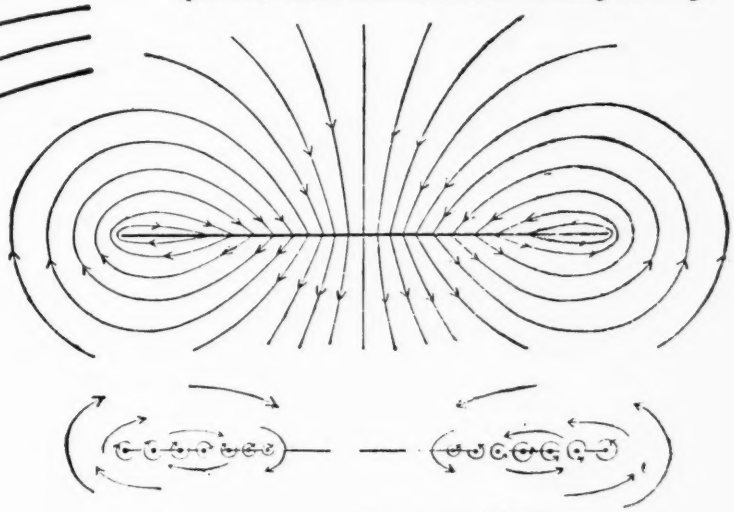


FIGURE 10 (*above*)

FIGURES 9-12. Vortex trunks. (*Aerodynamics*, Lanchester, Third Edition, 1911.)

Although Lanchester's "Aerodynamics" does not contain theoretically deduced quantitative rules for the influence of aspect ratio on pressure distribution, he presents a very interesting discussion of the forces acting on a rectangular flat plate as a function of the angle of attack and aspect ratio. He considers the rectangular flat plate in "Apteroid" and "Pterigoid" aspects. In the first case the longer side of the rectangle is in the flight direction and the aspect ratio is consequently less than unity; the second case corresponds to aspect ratios greater than unity, *i.e.* the longer side is perpendicular to the flight direction.

He concludes correctly for the Apteroid aspect that for the infinite lamina (zero aspect ratio) the sine-square law applies. In fact, in a non-viscid fluid this case is identical to that of an infinite lamina exposed to a velocity equal to $U \sin \alpha$ ($\alpha$=angle of inclination) and therefore the force normal to the lamina is proportional to $U^2 \sin^2 \alpha$. Lanchester makes the humorous remark, that "in the limiting case, which we may visualise as that of the broom-handle on which, it is alleged, in past times, witches were wont to perform their nocturnal flights, the law of the Newtonian medium applies." For the Pterigoid aspect he says that Rayleigh's formula based on the Kirchhoff wake theory gives values which are much too small, that the force is determined by the circulation but that the magnitude of the circulation is undetermined. It is evident that Lanchester did not then know of the work of Kutta and Joukowski and he did not adopt the so-called Kutta-Joukowski condition of smooth flow at the trailing edge, which plays an important role in present wing theory.

Eight years later, in 1915, Lanchester presented two papers to the Institution of Automobile Engineers.

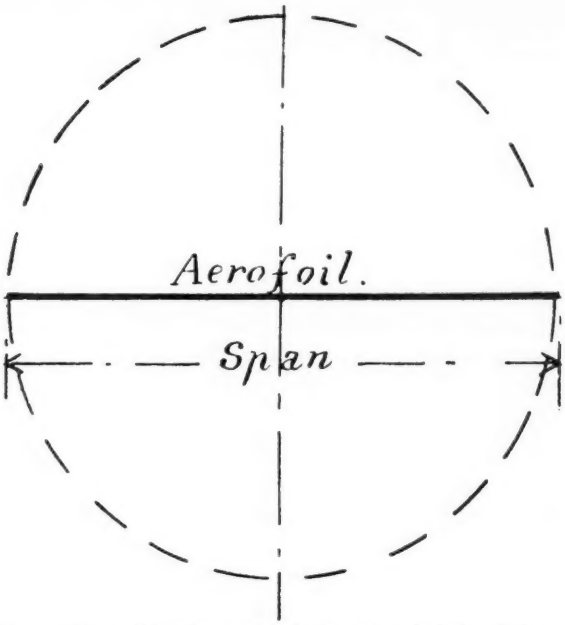



FIGURE 14. Fluid volume used in Lanchester's induced drag calculation. (*The Flying Machine—The Aerofoil and the Screw Propeller*, Lanchester, 1915.)

These papers also appeared in book form under the title "The Flying Machine—The Aerofoil and the Screw Propeller." In the first paper he gives the method of calculating the minimum aerodynamic drag unavoidably connected with the sustentation furnished by an aerofoil of finite span. He clearly states that, at least for wings of not too small aspect ratio, the energy aerodynamically necessary depends upon the span alone (inversely proportional to the square of the span) and is independent of the chordwise dimension.

His proposed calculation is based on the assumption that the lift is equal to the momentum of the downward motion of a fluid mass enclosed in a circular cylinder whose diameter is equal to the span of the aerofoil, as shown in Fig. 14, and whose length is equal to the length of the flight path in unit time. Then the kinetic energy of the same fluid mass has to be equal to the minimum work which must be spent in unit time to create such lift. The work is evidently equal to the product of the flight speed and the minimum aerodynamic drag (now called minimum induced drag). Lanchester does not state the formula but gives a sample calculation, which shows that he used, for the minimum aerodynamic resistance, the formula

$$R_{\min} = \frac{2}{\pi} \frac{L^2}{\rho U^2 l^2}$$

where $l$ is the span, $U$ is the flight speed, $L$ denotes the lift and $\rho$ the density of the fluid.

This formula is found for the first time, I believe, in a paper of A. Betz published in 1914. In the English translation Betz states that according to a recent theoretical investigation by Prandtl, which was to be published in the near future, it appears that the ratio between drag and lift for a given amount of lift and a given span cannot be less than a theoretical limit which is given by the formula

$$\left(\frac{R}{L}\right)_{\min.} = \frac{2}{\pi} \zeta L \frac{F}{l^2}$$

where $F$ is the surface area of the aerofoil. Since $L = \zeta_L F \rho U^2$, the two equations are identical.

How far Prandtl's work at Göttingen had been influenced by Lanchester's ideas in the development of his own theories was an often discussed question. I was a graduate student at the University when Lanchester visited Göttingen for the first time. Lanchester had much closer relations with Carl Runge, Prandtl's colleague teaching applied mathematics, for the reason that Runge spoke perfect English and Prandtl had difficulty expressing himself in that language. The connection between Lanchester and the Göttingen circle began in early 1908, according to Runge's daughter, Iris, who in 1949 published a biography of her father.* Runge was so impressed by the "Aerodynamics" that he proposed to translate Lanchester's book into German. Lanchester came to Göttingen to discuss details of the German translation and soon a very close friendship developed between the mathematician and

**Carl Runge und Sein Wissenschaftliches Werk*, Vandenhoeck and Ruprecht, Goettingen, 1949.

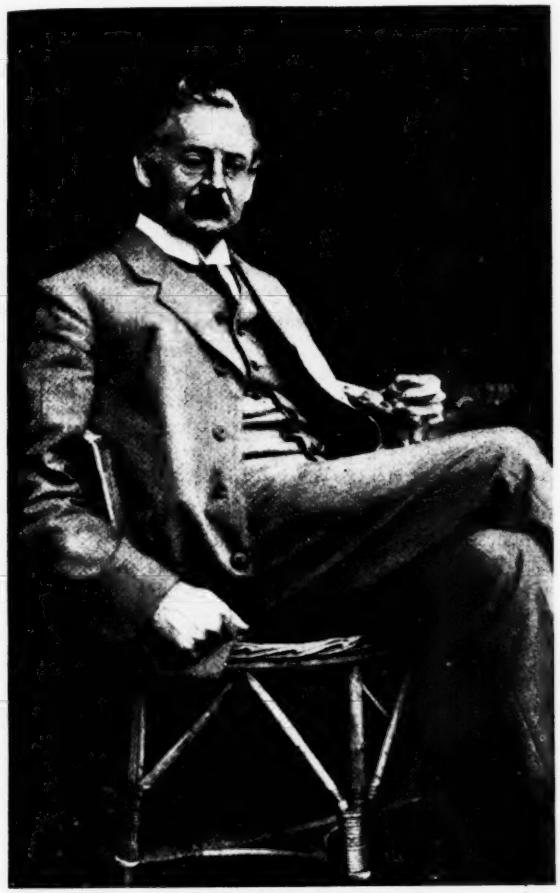


FIGURE 15. Carl Runge, 1912. A photograph taken by F. W. Lanchester and reproduced from *Carl Runge und Sein Wissenschaftliches Werk*, 1949.

the engineer, a friendship which survived the First World War of 1914-1918.

Prandtl himself, in his Wilbur Wright lecture in 1927, made the following remarks on the relationship between his and Lanchester's theories: "In England you refer to it as the Lanchester-Prandtl theory, and quite rightly so, because Lanchester obtained independently an important part of the results. He commenced working on the subject before I did, and this no doubt led people to believe that Lanchester's investigations, as set out in 1907 in his "Aerodynamics," led me to the ideas upon which the aerofoil theory was based. But this was not the case. The necessary ideas upon which to build up that theory, so far as these ideas are comprised in Lanchester's book, had already occurred to me before I saw the book. In support of this statement, I should like to point out that as a matter of fact we in Germany were better able to understand Lanchester's book when it appeared than you in England. English scientific men, indeed, have been reproached for the fact that they paid no attention to the theories expounded by their own countryman, whereas the Germans studied them closely and derived considerable benefit therefrom. The truth of the matter, however, is that Lanchester's treatment is difficult to follow, since it makes a very great demand on the reader's intuitive perceptions, and only because we had been working on similar lines were we able to grasp Lanchester's meaning at once. At the same time, however, I wish to be distinctly understood that in many particular respects Lanchester worked on different lines than we did, lines which were new to us, and that we were able to draw many useful ideas from his book."

Prandtl's lifting line theory which, with the possible exception of his boundary layer theory, is perhaps his most important and most applied contribution to theoretical fluid mechanics or aerodynamics, did not spring from its creator's head in one instant either. It was the result of long striving and many discussions. Even after the fundamental physical ideas were fixed, there were difficulties of a mathematical nature. In his report prepared for the National Advisory Committee for Aeronautics in 1925, Prandtl wrote: "For a long time it was difficult to find available functions to express the distribution of lift, from which a plausible distribution of the downwash would be obtained."

I remember that coming from Aachen I visited Prandtl, I believe in 1914. He complained of not having yet found a correct solution for the spanwise lift distribution. He tried a constant lift distribution first and obtained, of course, infinite downwash at the wing tips and an infinite value for the induced drag. Then he reasoned that the trouble resulted from starting with a finite lift per unit span at the tips. He therefore computed the drag for the case of a triangular distribution, but the result was again disappointing. I believe the correct solution, with lift per unit span increasing as the square root of the distance from the tip, was first published in 1918.

The second volume of Lanchester's "Aerial Flight," entitled "Aerodonetics," appeared in 1908. It contains detailed descriptions and computations on the motion of "aerodromes" (aeroplanes) and "aerodones" (gliders) especially from the viewpoint of possible flight paths and their stability. The best known chapters of this volume of 433 pages are those dealing with the phugoid motion. The phugoid is the motion of a statically stable aeroplane with a very small moment of inertia and a vanishing drag. It is assumed that such an aeroplane at every point of its flight path has the same constant attitude relative to the flight path (the curve described by its centre of gravity) and the only forces acting on the planes are gravity and a lift normal to the tangent of the flight path and proportional to the square of the speed of the centre of gravity. It is rather easy to compute the motion of such a system, but the picture of the family of possible trajectories, shown in Fig. 16, was a source of inspiration to the engineer.

Lanchester was not quite correct in the choice of the expression "phugoid." This is a somewhat amusing matter, since Lanchester often discussed the importance of a proper choice of aeronautical terms. He combined the two Greek words $\phi\upsilon\gamma\eta$ and $\varepsilon\iota\delta o\varsigma$ so that the combination would mean "flight-like." Now the second word has the meaning of likeness but unfortunately $\phi\upsilon\gamma\eta$

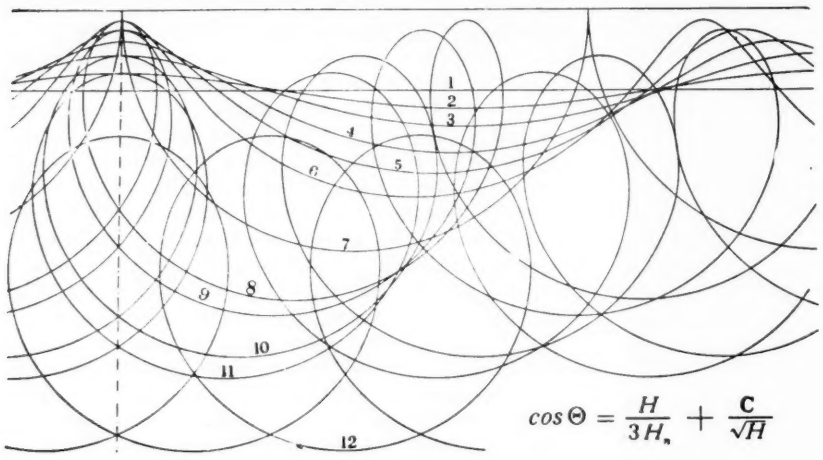



FIGURE 16. The Phugoid chart. (*Aerodonetics*, Lanchester, 1908.)

means flight only in the sense of "fleeing," *e.g.* running away from the battlefield, and has nothing to do with "flying," as a bird flies. Lanchester himself was aware of this distinction and conceded that the appropriateness of his derivation was "perhaps" diminished by this fact.

Nevertheless, one must admit that Lanchester invented quite a number of technical terms, using words of Greek origin, which became generally used and are also philologically correct. Few people are aware, however, that he invented the word "aerofoil" (or "airfoil" in the United States) composed of the Greek words ἀέρος and φυλλον meaning "air-leaf." Also the denomination of the early aeroplane of Samuel Franklin Cody as "Cathedral" goes back to Lanchester. It is generally supposed that this nickname was due to the large size of the aeroplane. As a matter of fact, however, it was due to a *bon mot* by Lanchester; as Cody was explaining that his aeroplanes had a slight negative dihedral, Lanchester remarked, "So you have a kat-hedral."

"Aerodonetics" also contains an interesting discussion of the longitudinal dynamic stability of wing and tail combinations. Lanchester used rather non-conventional mathematical methods (as Sir Melvill Jones put it); so much so that, for example, it took me several days to understand his deduction of the same stability condition which Bryan gave in his book "Stability in Aeronautics" in 1911 and Trefftz and I gave in 1914 as a first approximation. However, with an uncanny insight into the mechanics of the problem, Lanchester avoided many of the pitfalls which are usually connected with non-conventional mathematical methods. It is remarkable that he foresaw many points in the stability theory shown later by mathematical methods, for example, aeroelastic effects decreasing the stability of wings due to flexibility.

Using his own methods, Lanchester carried out a comparative calculation of the measure of stability for a series of examples starting with mica aerodones (glider models) weighing 0·25 to 4·0 grams and continuing with his own glider model, tried in 1894; the common swift (*hirundo apus*), illustrated in Fig. 17; and the wandering albatross (*diomedea exulans*) weighing 15 pounds, shown in Fig. 18. Finally he found that Lilienthal's glider, pictured in Fig. 19, with a total weight of 220 pounds, was unstable, at least below a speed of 30 m.p.h. His main conclusion was that a certain minimum velocity is necessary for stability. For example, a machine of three tons total weight would require flight at a velocity greater than 50 m.p.h. He therefore anticipated that the flying machine would not, according to the most sanguine estimate, exceed some few tons in weight unless a considerable advance was made in the prime mover.

The same—I would say almost conservative—spirit

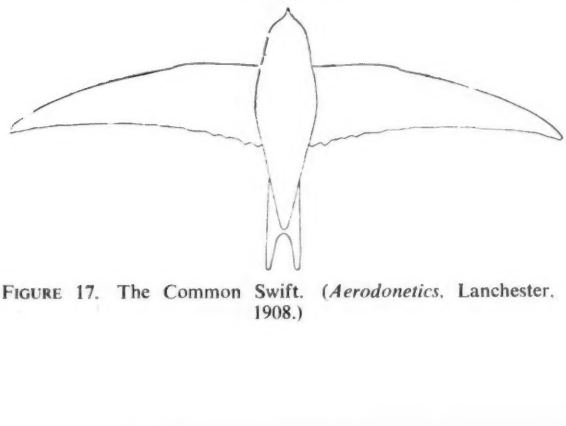


FIGURE 17. The Common Swift. (*Aerodonetics*, Lanchester, 1908.)

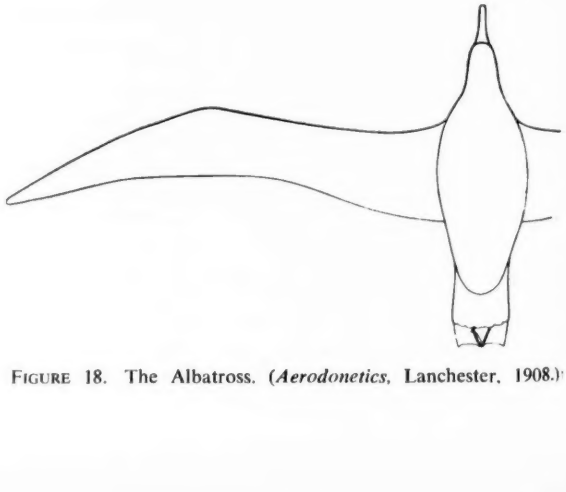


FIGURE 18. The Albatross. (*Aerodonetics*, Lanchester, 1908.)

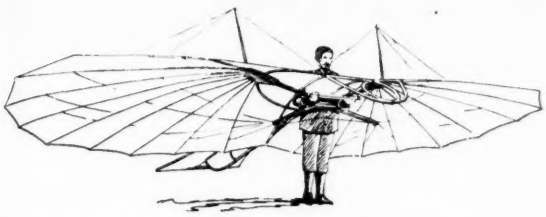


FIGURE 19. Lilienthal's glider. (*Aerodonetics*, Lanchester, 1908.)

is found in other predictions of Lanchester; for example, those concerning the future performance of aircraft. His "Aerodynamics" contains some modest estimates of possible ranges. Also his excellent James Forrest Lecture of 1915, published in book form in 1916 under the title "The Flying Machine from an Engineering Standpoint," gives testimony of a very sober way of looking into the future. In it he says: "We are so accustomed today to consider aerial flight as an essentially rapid mode of locomotion, that it is necessary to emphasize the fact that speed is obtained, not, as might readily be supposed, by virtue of any show of economy—as is true of the large steamship or railway train—but in spite of a coefficient of resistance and an expenditure of power of an altogether extravagant character." In 1907 he found that about 360 miles was an upper limit of the range of an aeroplane. In 1915 he found that, with a fuel quantity equal to 10 per cent of the mean weight, a range of 600 miles could be attained but that if it were considered fair "to take about one-third of the total (mean) weight as reasonably representing the full charge of petrol for a machine specially designed for long-distance flying, on this basis we have the possible range of flight represented by a distance of 2,000 miles, or, in the language of the Services, the radius of action is 1,000 miles."

It is also interesting to read the remark that "The direction in which advance is necessary before any great increase of speed can be realized is that of reduction in weight per horse-power of the motor; it is difficult to believe that any great reduction in weight can be effected on the best figure available today (about three pounds per b.h.p.) without an undue sacrifice of reliability."

In the same lecture Lanchester urged that not less than some 100 or 150 acres should be sacrificed for every important flight-ground. "It might be thought that the setting apart as flight-grounds of such considerable areas of land as above indicated would impose too serious a financial burden on flying, at least for some time to come, to be commercially possible. It is, however, to be borne in mind that with proper management such grounds could, especially if duplicated, be utilised for grazing purposes: thus, if an area of 200 acres were available, a herd of some few hundred head of cattle could be grazed, being transferred from one section of the ground to another from time to time. It is therefore evident that, under favourable conditions, the commercial aspect of the problem is by no means outrageous." To the best of my knowledge, Idlewild Airport occupies 5,070 acres but has no cattle.

The following predictions certainly stood the test of time:

"In offering to the public the first instalment of the present work, the author desires to record his conviction that the time is near when the study of Aerial Flight will take its place as one of the foremost of the applied sciences, one of which the underlying principles furnish some of the most beautiful and fascinating problems in the whole domain of practical dynamics."

"In order that real and consistent progress should be made in Aerodynamics and Aerodonetics, apart from their application in the engineering problem of mechanical flight, it is desirable, if not essential, that provision should be made for the special and systematic study of these subjects in one or more of our great Universities, provision in the form of an adequate endowment with proper scope for its employment under an effective and enlightened administration."

"The importance of this matter entitles it to rank almost as a National obligation; for the country in which facilities are given for the proper theoretical and experimental study of flight will inevitably find itself in the best position to take the lead in its application and practical development. That this must be considered a vital question from a National point of view is beyond dispute; under the conditions of the near future the command of the air must become at least as essential to the safety of the Empire as will be our continued supremacy on the high seas."

It is not possible for me to give you a full picture of the Lanchester approaches to many important questions which arose in the decades during which he followed the development of aviation. I would like to mention, however, a few of the papers or discussions he presented to various societies and technical journals. Two papers entitled "At High Altitude," printed in the JOURNAL of the Royal Aeronautical Society (May and June) in 1937, represent a very competent discussion of the correlation between high speed and high altitude with very much foresight into the development which took place in the following twenty years. A series of interesting short articles and discussions entitled "Rocket Efficiency" and "Exhaust Efflux Propulsion," of the period 1939-1941, deal with questions of propulsion by reaction. The paper "The Part Played by Skin Friction in Aeronautics" (1936)* is also worth studying. It reviews many subjects which Lanchester treated in previous publications. He refers, for example, to his earlier dispute with S. P. Langley, who believed that skin friction is relatively negligible and arrived at the conclusion that the power required for level flight decreases with the speed. Lanchester showed—what is now generally known—that the power required reaches a minimum at a certain speed.

Let us, however, return to the early period of the First World War. Lanchester's book with the title "Aircraft in Warfare" originated from a series of articles published in "Engineering" from September to December of 1914. It became one of the fundamental sources

**Journal of the Royal Aeronautical Society*, February and April 1937.

FIGURE 20. Air balloon engagement (an old print).

of inspiration for the science of Operational Analysis, which developed at a fast pace during the Second World War. Lanchester's book is devoted to the science of aircraft operation and war in the air. The idea of air warfare goes back to the early times of balloon flight. The inscription of a famous print from the year 1783 with the title "An Air Balloon Engagement for the Empire of the Sky," reproduced in Fig. 20, reads:

"As it is right
For the ocean to fight
Why should we not fight for the Air."

The print shows a French and a British hero, each tied by a rope to a balloon, fighting each other with swords. The real trial came in 1914.

Lanchester discusses in his book a great number of problems connected with aerial warfare: for example, strategic and tactical use of the Aeronautical (or fourth) Arm, armour and armament in the aircraft, attack by aeroplane and aeroplane bombing, rockets, airborne torpedoes, aeroplane *vs.* submarine, the ocean-going floating base (aircraft carrier), the fourth arm in peace time, the policy concerning government *vs.* private manufacture, and the like. The preface to the book is written by Major General David Henderson, a prominent expert on military aeronautics at that time. "During the past three years," he writes, "Mr. Lanchester and I have had several tussles in private on the questions debated in this book. Each can put up a pretty good defence on his own ground. Mr. Lanchester is well protected by his profound knowledge of physical science and his practical acquaintance with several branches of engineering. I am strongly entrenched behind a barricade of military prejudice, with some dim recollections of early scientific training as reserves for counter attack. In my incursions into Mr. Lanchester's territory I have now and then received a buffet which has made me more wary. And occasionally, I think, Mr. Lanchester has found himself hung up in my wire entanglements." I think military leaders and certain engineers and men of science who have to deal with the military will appreciate this quotation.

The outstanding feature, however, in "Aircraft in Warfare" (as was the phugoid theory in "Aerodonetics") which became well known and is considered as a contribution of great value to the fundamentals of Operational Analysis is Lanchester's "*N*-square Law." This law considers the value of numerical strength in weapons or weapon systems necessary for the annihilation of the enemy. It may be applied to the simplest weapon system, like man with gun *vs.* man with gun, or to more sophisticated weapon systems like battleships or aeroplanes *vs.* aeroplanes, or aeroplanes *vs.* ground defence systems.

The problem is in general as follows. Assume that the efficiency of the individual weapons of the two opposing parties is proportional to the numbers $\eta_1$ and $\eta_2$. For example, in the case of two opposing forces using guns, $\eta$ is proportional to the kill probability which a single fighting man can achieve in continuous firing. Let us denote the number of guns put into the fight by $N_1$ and $N_2$ for the two combatants, respectively. Lanchester shows that the fighting strengths of the two forces are equal when the squares of the numerical strengths multiplied by the fighting values of the individual units are equal, *i.e.*

$$\eta_1 N_1^2 = \eta_2 N_2^2.$$

One can easily interpret this rule. First, it is evident that the total fighting strength increases with the quality of the weapon and with the number of the weapons. Then, one must consider that the purpose is to wipe out the total strength of the enemy; hence, the effect of every kill on the enemy's strength is inversely proportional to his initial strength. If he has one hundred weapons, his relative loss is ten times smaller than if he has one thousand weapons. Therefore the formula for equivalence is

$$\eta_1 \frac{N_1}{N_2} = \eta_2 \frac{N_2}{N_1}$$

which is another way of expressing the *N*-square Law.

What are the obvious conclusions from the *N*-square Law? First, one has to say that if the enemy has weapon systems twice as efficient as yours, his advantage can only be balanced by increasing your numerical

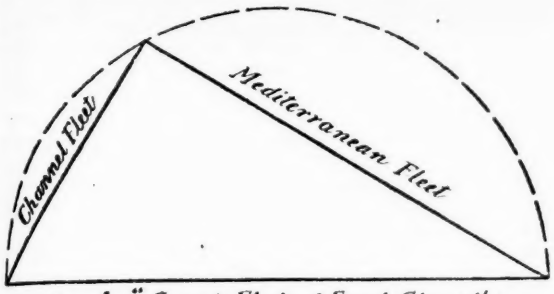



FIGURE 21. The *N*-square Law. (*Aircraft in Warfare*, Lanchester, 1916.)

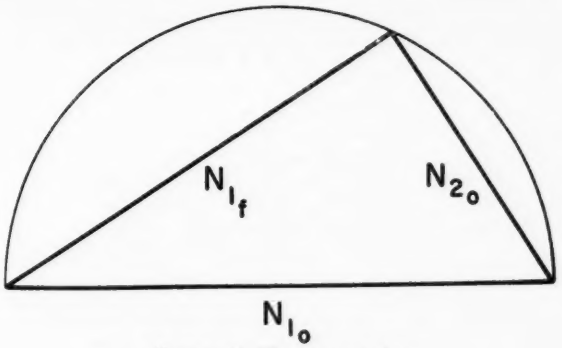



FIGURE 22. The *N*-square Law.

strength by a factor of four. Second, if the quality is assumed to be the same, the only help against the enemy's total numerical superiority is the principle of concentration. In this special case it is easy to show that if initially $N_{1_0} > N_{2_0}$, during the fight the difference $N_1^2 - N_2^2$ remains constant ($N_1$ and $N_2$ are the numbers of the remaining weapon systems). Hence, when force 2 is totally wiped out, the remainder of force 1 is equal to $\sqrt{(N_{1_0}^2 - N_{2_0}^2)}$. This relation is shown in Fig. 22. Obviously $\sqrt{(N_{1_0}^2 - N_{2_0}^2)}$ is the third side of a right triangle whose hypotenuse has the length $N_{1_0}$ and whose other side is equal to $N_{2_0}$.

Thus, for example, if you and your adversary have equal total forces but you succeed in concentrating your forces on the two halves of the enemy forces consecutively, you suffer only about a 30 per cent loss of your forces when the enemy force is totally wiped out. If you attack at any time only one fourth of the total enemy force, after three encounters you come out even against an enemy who initially had double the strength you had. The enemy lost 75 per cent of his force, whereas you lost only 50 per cent.

Lanchester quotes quite a number of historical examples which show the principle of concentration expressed by the *N*-square Law. Thus he analyses Nelson's tactical scheme employed in 1805, in the battle of Trafalgar. He believes that the Nelson "Touch" essentially consisted of an intuitive application of the *N*-square Law. Lanchester's law is given in almost any textbook for Operational Analysis and has been further developed by a number of authors.

This is the closing argument for my review of Lanchester's contributions to aeronautical science and the science of air warfare. I do not wish to review his many papers dealing with a long list of other scientific and technological subjects. As an example of the broad extent of his interest, I may mention that in 1935 he published a book with the title "Relativity, an Elementary Explanation of the Space-time Relations as Established by Minkowski and a Discussion of Gravitational Theory Based Thereon." The book is dedicated to the memory of his close friend Carl Runge of Göttingen. According to Iris Runge, whose biography of her father we have already quoted, Lanchester was quite worried about the theory of relativity and discussed the subject orally and in correspondence with his friend Runge. According to this source, Runge succeeded in eliminating most of the logical objections of Lanchester to the relativistic philosophy of space and time. Lanchester also wrote a very remarkable book on the theory of dimensions (1935); he dealt especially with the engineering praxis using the "slug" and was also critical about the use of electro-magnetical units.

Finally, a casual personal remembrance: After I met Lanchester several times in Göttingen in 1908, I saw him in Cambridge in 1911 on the occasion of the Fifth International Congress for Mathematics where I made a short contribution concerning the vortex trail "whose name I have the honor to wear." Lanchester made a point of showing me the sights not only in Cambridge but in the environs; for example, the Cathedral of Ely. He drove an automobile with a speed which appeared to me excessive in view of the rather narrow, winding country roads. We discussed aerodynamics. I was at that time more used to riding in one-horse buggy-cars than in automobiles, and meekly expressed my worry about his speed of driving. He looked at me and only said, "I am surprised; I thought your name was car-man."

Summarising this study of Lanchester's contributions, we have to admire his physical insight into the fundamentals of aerodynamics and flight mechanics, his enthusiasm to serve the cause of Aeronautics and Air Power. We may regret that he did not enjoy the full recognition of his contributions as he deserved.

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SIR GEORGE EDWARDS: Thank you very much, Sir, for a magnificent address. I think I ought to put it on record that the Society to which you referred "in some modern time, such as 1870, something that was a forerunner of this Society." Stamped all over my badge of office loud and clear, although I am looking at it upsidedown, it says 1866. So it was really this Society.

Now it is the custom at these lectures not to have a discussion and I will call on Dr. Eric Moult, who is Vice-President of the Society, to propose the vote of thanks to Dr. von Kármán.

DR. E. S. MOULT, F.R.Ae.S.: Mr. President, Dr. von Kármán, ladies and gentlemen. We are here tonight to pay tribute to the memory of a British pioneer in the science of aeronautics, to the name of Lanchester. As a Society, indeed as a Nation, we are privileged in that this tribute has been given by a man who himself has been described as "the outstanding aeronautical scientist of our generation." Theodore von Kármán is an international figure in international aeronautics. His fame and standing are equally recognised on both sides of the Atlantic. He founded and established the Aeronautical Institute of the University of Aachen: for a period he was at the same time the Director of the Guggenheim Laboratory of the California Institute of Technology. Thus we see that in effect our Lecturer was simultaneously the Professor of two Universities, one on each side of the Atlantic. Such, ladies and gentlemen, is the stature of the man who has addressed us tonight.

Dr. von Kármán has the good fortune to live at Pasadena in California. There he has organised and become the Director of the Jet Propulsion Laboratory. Some of us, two years ago, had the opportunity of visiting that Laboratory and seeing something of its work. We were impressed by the work in hand and by the facilities that were available, but I think we were more impressed by the spirit of veneration and affection which the staff of that establishment and the students bore for Dr. von Kármán. Wherever he goes, Dr. von Kármán gives an inspiration to scientific work and its practical application. In this country he is perhaps best known for his works on fluid motion, but his writings I find cover a vast range of subjects: Aerodynamics, Thermodynamics, Combustion, Strength of Materials, Structures, altogether an extraordinary breadth and variety of subjects and interests. In his writings we have a clarity, a conciseness and a practical application which we all agree is characteristic of the great mind. Sir, we are most grateful to you for coming to England and giving us this address tonight. It is indeed a great occasion and it has been most inspiring. Any words of mine are quite inadequate to do justice and with the greatest humility I say on behalf of us all, thank you very much indeed.

PROFESSOR A. D. BAXTER, F.R.Ae.S. (Member of Council): It is difficult for me to say more than Dr. Moult has already said in proposing the vote of thanks and it is perhaps hardly appropriate that ordinary mortals like myself should suggest complimenting or congratulating such a master as Dr. von Kármán on his exposition of Lanchester this evening.

He is not merely a master—I was almost tempted to say a past master at lecturing, but that is not true either because he is very much in the present. I think perhaps his real title is *the* master of aeronautics, and the only tribute that is necessary tonight has been paid already; that is, in the size of the audience. I can assure you that it is not every lecturer before this Society who is able to fill this hall till the walls are literally bulging.

It has not been merely a memorial lecture this evening. I am sure that one could say it has been equally a memorable lecture. Merely to have the almost legendary figure of von Kármán present at one of our meetings is an occasion to remember, but to have him actually speaking to us as we have this evening with all the wisdom of his long years of experience is something which is very outstanding. I feel that tonight will impress itself on the minds and memories of most of us, especially perhaps on the younger members. Just as some of our more senior members are apt to boast of the occasions when they met some of the early pioneers of aeronautics, so, I am sure, in 30 or 40 years or even perhaps when we come to the celebration of one hundred years of powered flight, there will be some of the people who are in this room this evening who will be laying, as one of their claims to fame, the fact that they were here and heard the great von Kármán give the first Lanchester Memorial Lecture. I am sure that they will still appreciate some of the things that have been said tonight. I am sure we all do now and we all feel something of the personality that has brought students to him from all over the world, something of the genius that has, as Dr. Moult has said, built up two great research institutes to a world reputation and founded several international organisations, and something of the inspiration with which he encouraged his colleagues in many ways.

To say more would be attempting to "gild the lily." All I would like to add is that I am certain that all of us in this hall have gained something by being present here tonight. I am certain that we all realise the privilege we have had and the honour you, Dr. von Kármán, have done to our Society. We do desire to thank you most sincerely on this occasion and, if I might add to what the President said earlier, we would like to congratulate you on the vigour that you display on this eve of your 76th birthday. We would like to wish you a very happy birthday and very many still to come in which to pursue your activities.

It is twenty years almost exactly since Dr. von Kármán gave his Wilbur Wright Lecture to the Society. He was described at that time as an eminent American. This evening we have had another lecture from him and he is described as an international figure. I am tempted to wonder whether we will be looking for a lecture from him in another twenty years and whether at that time he will be described as an interplanetary figure. I am certain, anyhow, that if he were to give us a lecture then it would be just as clear and capable, just as full of humour and as enjoyable to all of us as the one he has given us tonight.

I have very great pleasure indeed in seconding this vote of thanks.

SIR GEORGE EDWARDS: We have listened not only to a magnificent lecture, but if I might say so to two excellent votes of thanks.

Before I put the vote of thanks to the meeting, if I might bring your minds back to Lanchester for a moment, it does seem both from what Dr. von Kármán and others have said that this great Englishman was not really fully recognised for the job that he did. A lot of it he did too soon before the people were ready to absorb it and it has been said that his work was not honoured as it should have been. I would like to read to you for two minutes an extract from the Obituary Notice published by the Royal Society (Vol. 5, Part 16) written by Harry Ricardo at the time of Lanchester's death. (Laurence Pritchard, who knew him well, said to me that it gives the best picture of the man that we can conceive in a short space.) He said this:—

> "In Lanchester we had one of the very rare combinations of a great scientist, a great engineer, a mathematician, an inventor and a true artist in mechanical design. He was a poet and a philosopher. His mind worked so quickly and so directly through the intermediate stages of thought that few could keep pace with it and this rendered him rather intolerant of their slower processes. For this reason he was a difficult man to help and nearly all the work he

achieved, he achieved single-handed. Whatever problem he tackled, he tackled always from first principles for he refused to be influenced by fashion or the belief of others unless they accorded with his own experience or common sense. He had a very fine sense of humour which sometimes he employed a little mercilessly for he could not tolerate the superficial or the second rate. Like all great inventors who invent ahead of their time Lanchester had to endure the mortification of seeing his own inventions reappear years later perhaps in a slightly modified form and under another name. This he took very hardly for he failed to recognise and accept that this inevitably is the fate of all such."

And if it is so that Lanchester was not treated as well as he should have been treated, and if he has not been honoured as he should have been, then I hope that tonight with this address that has been given by Dr. von Kármán, and by the attendance of these people—nearly all young ones—I hope that this will be a demonstration that at least we in the Aeronautical Society believe that he was the greatest English aerodynamic pioneer.

And if I might now say to you, Madam, who as his wife sustained him through his years of work and endeavour, and to you, Sir, his brother, that if what has happened tonight has done something to convey to you the thought that here was a man who needed to be honoured, then Dr. von Kármán and the rest of us will have achieved what we set out to do. May I say further how delighted we all are that you should come tonight.

And now ladies and gentlemen, the vote of thanks to Dr. von Kármán has been proposed and seconded and I would submit it to you for your approval.

And if I may thank you all very much for making this such a great occasion by such a large audience I will close the meeting.

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Following the lecture a dinner was given by the President and Council at 4 Hamilton Place, London, W.1, at which the following were present:—

Mr. L. S. Armandias, A.F.R.Ae.S., Deputy Managing Director, British Messier.

Dr. A. M. Ballantyne, T.D., B.Sc.(Eng.), Hon.F.C.A.I., A.F.I.A.S., F.R.Ae.S., Secretary, Royal Aeronautical Society. Professor A. D. Baxter, M.Eng., M.I.Mech.E., F.R.Ae.S., Member of Council; Deputy Principal, College of Aeronautics, Cranfield. Professor P. M. S. Blackett, F.R.S., President, British Association for the Advancement of Science. Mr. E. C. Bowyer, C.B.E., Director and Chief Executive, Society of British Aircraft Constructors Ltd. The Lord Brabazon of Tara, P.C., G.B.E., M.C., Hon.F.R.Ae.S., President, Royal Aero Club; Past President, Royal Aeronautical Society. Major G. P. Bulman, C.B.E., B.Sc., F.R.Ae.S., Honorary Treasurer, Royal Aeronautical Society and Past President; Director of Construction of Research Facilities, Ministry of Supply.

Sir Sydney Camm, C.B.E., F.R.Ae.S., Past President, Royal Aeronautical Society; Director and Chief Designer, Hawker Aircraft Ltd. Professor A. R. Collar, M.A., D.Sc., A.F.I.A.S., F.R.Ae.S., Member of Council; Professor of Aeronautical Engineering, University of Bristol.

Sir George Edwards, C.B.E., B.Sc., F.R.Ae.S., President, Royal Aeronautical Society; Managing Director, Vickers-Armstrongs (Aircraft) Ltd.

Sir William S. Farren, C.B., M.B.E., F.R.S., M.I.Mech.E., Hon.F.I.A.S., F.R.Ae.S., Past President, Royal Aeronautical Society; Technical Director, A. V. Roe and Co. Ltd. Mr. L. G. Frise, B.Sc., A.F.I.A.S., F.R.Ae.S., Member of Council; Director of Special Projects, Blackburn and General Aircraft Ltd.

Mr. G. W. H. Gardner, C.B.E., B.Sc., F.R.Ae.S., Member of Council; Director, Royal Aircraft Establishment. Mr. H. H. Gardner, B.Sc., F.R.Ae.S., Member of Council; Director and Chief Engineer, Military Aircraft, Vickers-Armstrongs Ltd.

Sir Arnold A. Hall, M.A., F.R.S., F.R.Ae.S., Vice President, Royal Aeronautical Society; Technical Director, Hawker Siddeley Group. Mr. Stewart Scott Hall, C.B., M.Sc., D.I.C., F.C.G.I., F.I.A.S., F.R.Ae.S., Member of Council; Scientific Adviser, Air Ministry. Mr. P. A. Hearne, D.C.Ae., D.L.C., Grad.R.Ae.S., Helicopter Project Engineer, Project and Development Branch, British European Airways.

Mr. E. T. Jones, C.B., O.B.E., M.Eng., F.R.Ae.S., Past President, Royal Aeronautical Society; Director General of Technical Development (Air), Ministry of Supply. Air Marshal Sir Owen Jones, K.B.E., C.B., A.F.C., B.A., D.I.C., M.I.Mech.E., F.R.Ae.S., Member of Council; Aeronautical Consultant.

Dr. T. von Kármán, Hon.F.R.Ae.S., First Lanchester Memorial Lecturer; Chairman, Advisory Group for Aeronautical Research and Development of the North Atlantic Treaty Organisation.

Dr. G. V. Lachmann, F.R.Ae.S., Research Director, Handley Page Ltd. Mr. George Lanchester, Brother of the late Mr. F. W. Lanchester.

Professor W. A. Mair, M.A., A.F.I.A.S., F.R.Ae.S., Francis Mond Professor of Aeronautical Engineering, Cambridge University. Mr. P. G. Masefield, M.A., Hon.F.I.A.S., F.R.Ae.S., Vice President, Royal Aeronautical Society; Managing Director, Bristol Aircraft Ltd. Mr. M. B. Morgan, M.A., F.R.Ae.S., Member of Council; Deputy Director, Royal Aircraft Establishment. Dr. E. S. Moult, B.Sc., M.I.Mech.E., F.R.Ae.S., Vice President, Royal Aeronautical Society; Chief Engineer and Director, The de Havilland Engine Co. Ltd.

Mr. F. M. Owner, C.B.E., M.Sc.(Tech.), M.I.Mech.E., F.R.Ae.S., Member of Council; Deputy Chief Engineer, The de Havilland Engine Co. Ltd.

Colonel R. L. Preston, C.B.E., A.F.R.Ae.S., Secretary-General, Royal Aero Club. Captain J. L. Pritchard, C.B.E., Hon.F.I.A.S., Hon.F.R.Ae.S., Formerly Secretary, Royal Aeronautical Society.

Professor E. J. Richards, M.A., B.Sc., F.R.Ae.S., Professor of Aeronautical Engineering, University of Southampton. Mr. N. E. Rowe, C.B.E., B.Sc., F.C.G.I., F.I.A.S., F.R.Ae.S., Past President, Royal Aeronautical Society; Technical Director, Blackburn and General Aircraft Ltd.

Mr. B. S. Shenstone, M.A.Sc., F.C.A.I., A.F.I.A.S., F.R.Ae.S., Member of Council; Chief Engineer, British European Airways. Sir Richard V. Southwell, M.A., Hon.F.I.A.S., F.R.Ae.S.

Mr. C. F. Uwins, O.B.E., A.F.C., F.R.Ae.S., President, Society of British Aircraft Constructors Ltd.

Dr. P. B. Walker, C.B.E., M.A., F.R.Ae.S., Head of Structures Department, Royal Aircraft Establishment. Dr. F. L. Wattendorf, M.S., F.I.A.S., M.A.S.M.E., M.A.P.S., F.R.Ae.S., Director, Secretariat, AGARD. Mr. L. A. Wingfield, M.C., D.F.C., A.R.Ae.S., Solicitor to the Royal Aeronautical Society.

# Replacement of Piston Engines by Gas Turbines in Air Liners*

by

H. SAMMONS, C.B.E., M.I.Mech.E., F.R.Ae.S.

(*Managing Director, D. Napier and Son Ltd.*)

## Introduction

From time to time, both military and civil aircraft have had a change of power plant to meet special circumstances but in all cases considerable redesign has been necessary and I know of no instance where it has been done in the field by means of a "modification kit." It has become quite commonplace to do so for test purposes, that is to make the aeroplane a flying test bed. Douglas DC-3 aircraft have been fitted with Darts and Mambas, Varsity with Elands, Canberras with Olympus and many others.

Some three years ago, much detailed consideration was given to the possibility of civil air liners in service having their piston-engined power plants replaced by turbo-propeller engines in the belief that it was both possible and desirable. It was obviously possible from an engineering standpoint as such changes had been done for flight test purposes, but much study had to be made on whether airline operators would be attracted by such a proposal.

It is evident that the operation of civil air liners is finely balanced economically and any adverse factor very easily and quickly makes a profitable operation into a losing one. Therefore, the conversion of an existing operating air liner is only justified if it does not cause a loss, but increases the earning power of the aircraft; otherwise, the operator would not be justified in making the conversion. This called for a thorough investigation and it became evident that many issues were involved. The primary requirement was that, for any given power plant, there must be sufficient aircraft in service to justify the production of the modification kits and indeed, in an extreme case, the engine itself.

The inherent advantages of gas turbine-powered aircraft are recognised by both airline operators and passengers and it is fairly safe to say that all future orders will be for this type of equipment.

At the present time the composition of the World fleets (excluding Russia and China) comprise 4,300 piston-engined aircraft and 170 gas turbine-powered aircraft. By 1960, it is estimated that the relative figures will be 3,900 piston-engined aircraft and 800 gas turbine-powered aircraft. By 1963, when all present orders for the more advanced types, including the large jets, will have been delivered the relative figures will be 3,100 piston engined and 1,500 gas turbine aircraft.

It will be seen then that, even in 1963, there will still be a preponderance of piston-engined aircraft. This is shown graphically in Fig. 1 where curves have also been added to show various percentage increases in capacity seat miles.

If it is assumed that the growth of air traffic will be between 15 and 20 per cent per annum, then the planned production of aircraft will be able to meet the demands, provided that, substantially, existing aircraft will remain in commercial operation.

Many of the remaining piston-engined types are presently operating at speeds well below their design level flight speed and have cabin pressurisation systems allowing flight at greater altitudes than they are presently operating, and it is these types which can be considered for conversion to propeller-turbine power and help to make up the deficiency of gas turbine-powered aircraft.

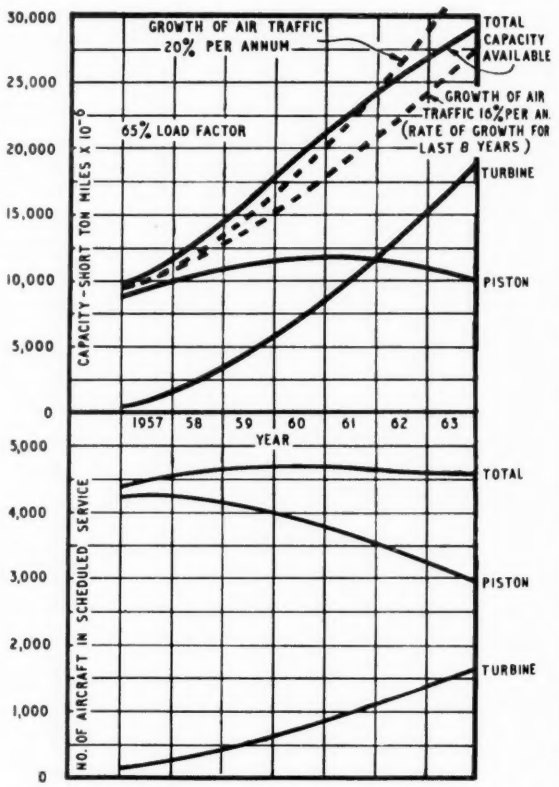


FIGURE 1. Estimated number of civil aircraft in scheduled service up to 1963.

*The Second Henson and Stringfellow Memorial Lecture of the Yeovil Branch of the Society, given at Yeovil on 21st February 1957.

FIGURE 2. Standard Convair 340 aircraft.

FIGURE 3. Turbine-engined Convair 340.

The following factors are of paramount importance in the consideration of a "conversion": —

(*a*) Cost of conversion.
(*b*) Cost of operation.
(*c*) Performance of converted aircraft.
(*d*) Loss of revenue during the conversion.
(*e*) Reliability of converted aircraft.

Items (*b*) and (*c*) must show to advantage over the piston-engined machine and (*e*) must not be inferior but, preferably, superior. Items (*a*) and (*d*) seriously influence (*b*), for if these items are unduly costly, then it makes it impossible to obtain a lower cost of operation.

A more elusive factor which has to be taken into account in the final assessment of the profit-earning relationship after conversion, is what the extra revenue-earning capacity will be, if any. Certainly, better performance by way of increased cruising speed will enable more passengers to be carried per year as more trips can be made and, the passengers will spend less time on the journey. Also, the turbine-engined aircraft gives more comfort to the passengers through reduction of noise and vibration. Therefore, in competition, the turbine-engined aircraft will attract and carry more passengers than a competitive piston-engined aircraft when operating on the same route. Theoretically, it would be possible to have an aircraft with higher direct operating costs and yet make a bigger profit, owing to its revenue being much increased. However, I hope to show that, in the cases considered, well-known airlines could operate at reduced costs and increased revenue by converting to turbo-propeller power plants.

Obviously, one can only choose for consideration those aeroplanes where the power plant is ideally suitable and, as the engine to be considered in this case was the Napier 3,500 h.p. Eland, of all the aeroplanes operating throughout the world the following types were found to be eminently suitable for conversion: —

(*a*) Convair 340 and 440.
(*b*) Douglas DC-6B.
(*c*) Constellation 749.

As there are in operation, or on order, at the present time 285 Convairs, 321 DC-6B and 325 Constellation 749, I make no apologies for choosing American aircraft for there are no comparable British piston-engined aircraft existing in sufficient quantities to justify conversion. Also, as the total aircraft given amounts to a potential of 4,730 engines, including spares, if only one-third of the aircraft were converted it is more than enough to justify the production of engines and modification kits, provided that the change was not too great or costly. Just as it is essential to have a period of development for a new aircraft, so it is for a "conversion," both for obtaining official certification and for demonstrating to intended purchasers.

A decision was made to convert a new twin-engined air liner of the latest type in operation and do the conversion as a sample; for this a standard Convair 340 was chosen as shown on Fig. 2. Concurrently, an Elizabethan air liner was also converted so as not to confine the experiment to one aircraft.

In considering the conversion of both these aircraft, it was necessary to decide how far to go in the redesign. Both had large diameter radial, air-cooled piston engines which meant that upon installing a turbo-propeller engine of approximately half the diameter, a reduction of the nacelle diameter by at least 12 in. was possible, which, on first thoughts, seemed to be desirable. There would obviously be a reduced drag and improved performance but it was decided not to do this because the modification brought into being changes to the wing, which was quite prohibitive from a cost point of view. It was found possible, in both aircraft, to make the change forward of the existing bulkhead and so reduce the cost of conversion to a minimum. I mention this as one of the factors which was determined mainly from an economical point of view, for it applied whether it be a twin or four-engined aircraft, as any change aft of the bulkhead materially increased the cost of the conversion without giving adequate improvement in performance.

As it is impossible in a paper of this length to give a detailed description of the conversion of all the types of aircraft considered, the work done on the Convair 340 will be described, including the assessment of the economics of its operation in comparison with the original piston-engined version. Comparisons are also given in less detail of typical four-engined air liners.

Figure 3 shows the Convair air liner after conversion to turbo-propeller engines.

Although it is probably unusual to enter into economics in detail in a technical lecture, it is impossible to treat this subject usefully without doing so. The paper, therefore, has been divided into three sections: —

1. Details of the changes in the aircraft.
2. The change in the performance of the aircraft.
3. A study of the economics of the aircraft, both before and after conversion.

## Engine Nacelles

It has already been stated that an early decision was made to retain the original nacelle diameter of approximately 66 in. Although it offends the eye to have such a large diameter, the subsequent estimation of the performance penalty proved that this decision was undoubtedly right, since the estimated loss in cruising speed resulting from the larger nacelle is only about two knots (2·303 m.p.h.).

Maintenance accessibility would also have been decreased if such a change had been made, and the propeller disc would have been brought forward some inches closer to the main entrance door. This is shown in Fig. 4, which is a comparison of the two nacelle lines.

The major structural changes actually embodied in the aircraft are as follows: —

(*a*) The nacelles were locally reinforced to accept the greater torque loads resulting from the installation of the higher powered turbine engines. This reinforcing consisted primarily of increasing the number of stringers between the nacelle frame, and changing the section of the longerons adjacent to the engine pick-up points by "boxing-in" the original channel sections. The areas adjacent to the new stringers were also re-skinned.

(*b*) Modified nacelle top structure and skin to permit the installation of the engine jet pipe and hot air muff. This was necessary in view of the difference between the single turbine jet pipe and the twin exhaust and thrust augmentor tubes of the original aircraft.

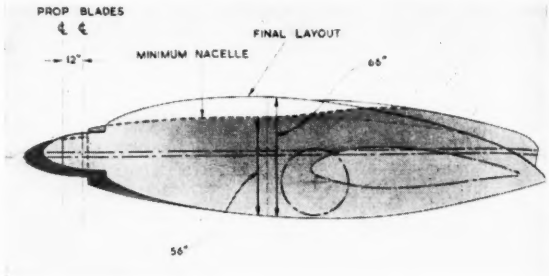


FIGURE 4. Comparison of minimum nacelle shape and the layout actually adopted with unchanged rear structure.

Consideration was given in the early design stages to the possibility of using an elliptical section jet pipe, in an attempt to reduce the extent of the changes to the nacelle profile. The additional stiffening required to give adequate strength to the pipe under differential pressure loading, plus the complication of the hot air muff, would have doubled the weight of the pipe, and it was therefore decided to adopt the circular section and accept the greater degree of modification to the nacelle.

(*c*) A new fire wall of greater structural stiffness was considered desirable, particularly in view of the size of cut-out required to enable the jet pipe to pass through, and the mounting of the accessory gearboxes which would rely principally upon the fire wall for their support. The existing engine mounting pick-up points were repositioned slightly, and the fittings were changed from light alloy to steel for greater strength.

## Airframe

The engine to be installed in the aircraft develops 3,500 e.s.h.p. having the usual maximum continuous and cruising ratings. To avoid increasing the areas of both vertical and horizontal tail surfaces of the standard Convair aircraft, to cater for the "engine failed" case on take-off, the power is torque-restricted 3,000 s.h.p. (3,350 e.s.h.p.).

By adopting this policy, it has been possible to eliminate airframe modifications and, at the same time, provide an economically satisfactory engine rating which has a reserve of power under high ambient temperature conditions. Further comment on this will

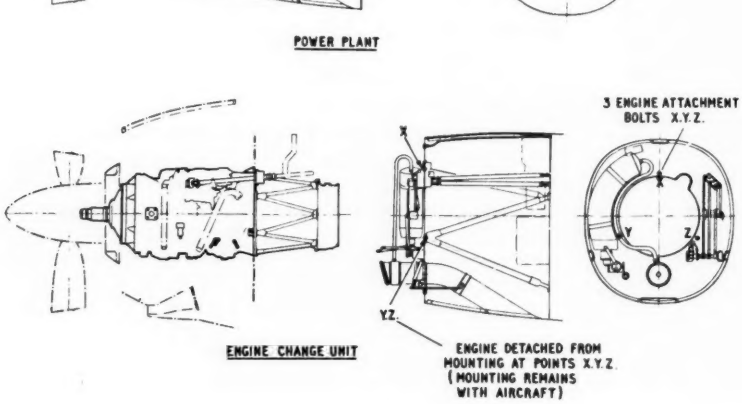


FIGURE 5. The engine or power plant change principle.

be given later when the performance of the aircraft is discussed.

## Engine Power Plant

The power plant has been designed to permit removal and re-installation as a complete power unit or to enable the engine only to be removed. This gives the operator freedom to select the best maintenance procedure to suit his particular operating conditions.

Where the aim is the maximum utilisation of the aircraft, it is better to remove and re-install a complete power unit, thus avoiding the necessity of removing and re-fitting such things as cowlings, oil system components, and so on. Where economy is paramount as, for example, with a small operator, the loss in utilisation may be countered by the saving in cost resulting from changing the engine only, since duplication of cowling panels, engine mounting, and so on, is avoided.

The principle of engine or power plant change is illustrated in Fig. 5. It is considered that power plant change could be accomplished in under an hour, whereas an engine change could be made over-night.

Full accessibility to all parts of the engine is provided by seven bonnet-type cowlings embodying quick-release toggle fasteners. In addition, the cowling rails are attached by means of pip-pins, which enables complete cowling assemblies to be rapidly removed, if necessary, for maintenance or component change.

The lower panel of the three compressor bay cowlings embodies the oil cooler intake duct. Four panels are used over the turbine bay due to the greater diameter and length. All cowlings and stays are stressed for 40 knot side-winds to permit maintenance on the open aerodrome, with the panels open.

The compressor and turbine bays are separated by an intermediate fire wall. No accessory equipment is contained in the turbine bay, other than fire detector and extinguishant spray rings; all items likely to be affected by heat, or to contribute to a fire, are contained in the compressor bay where no surface temperatures

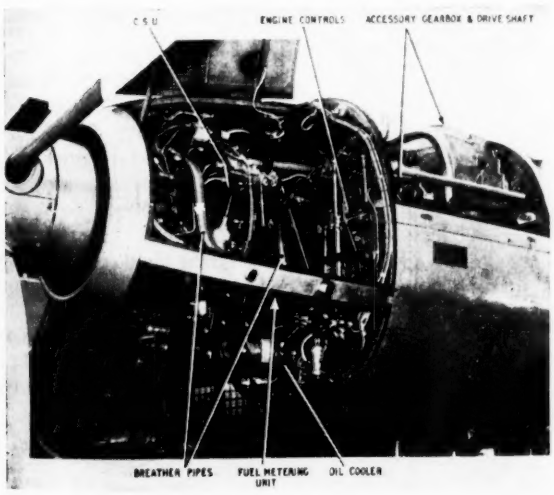


FIGURE 6. Arrangement of engine cowlings.

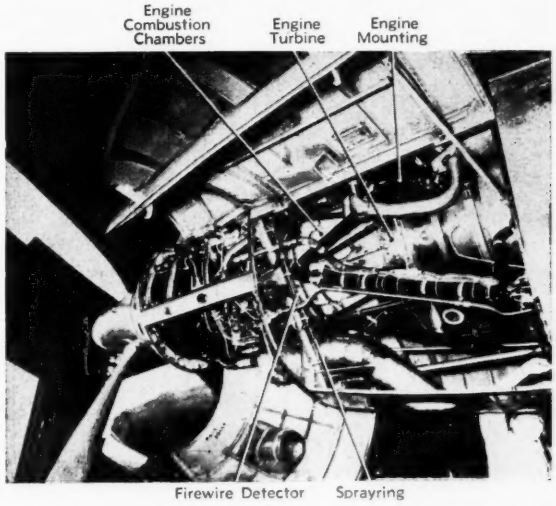


FIGURE 7. Arrangement of turbine bay components.

in excess of 200°C are ever attained. Both bays are adequately ventilated, the exit air louvres or outlets being positioned in a manner which will prevent excessive "soak" temperatures being reached after the engine has been shut down. Bay temperatures in flight are seldom above 60°C, and after protracted ground running, 120°C.

The engine lubricating oil system is contained entirely in the compressor bay and includes the oil tank, with a gross oil capacity of about seven gallons. This includes a propeller feathering oil reserve of 2½ gallons and provision for a minimum of five seconds oil supply under negative-*g* conditions.

An eight inch diameter, air-cooled, light alloy oil cooler is fitted, oil temperature control being effected through a butterfly valve in the exhaust duct. This is electrically-actuated and pilot-controlled, although thermostatic operation can be introduced if required.

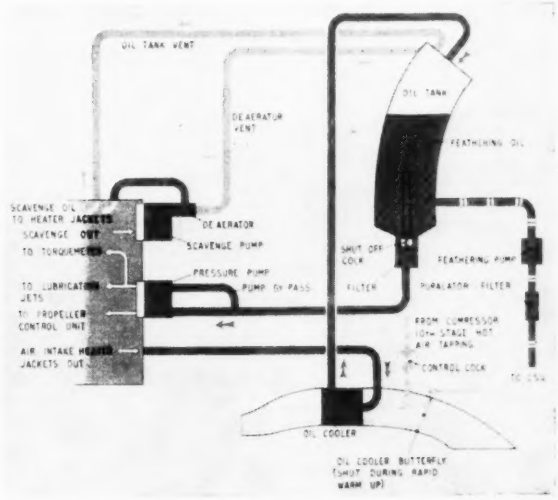


FIGURE 8. Oil system layout.

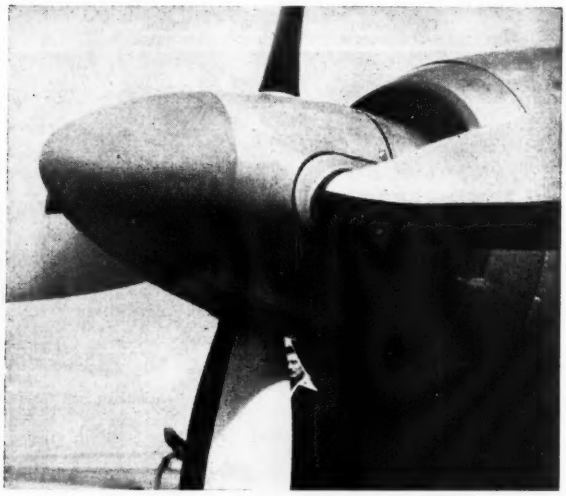

FIGURE 9. Anti-icing protection on engine intake and propeller.

To ensure immunity from icing, the engine air intake passages through the reduction gear are heated by circulating hot engine oil through jackets cast around the air passage. To enable the necessary oil temperature to be reached in the least possible time from starting the engine, a rapid warm-up system can be used whereby the oil cooler butterfly is closed and hot air from a compressor tapping is made to flow through the oil cooler air passages, thus heating the oil. This will ensure ice immunity at low temperatures within three or four minutes of starting the engine, and enable the operator to make full use of the advantages of the turbine engine, which does not basically require protracted warming-up before take-off, as piston engines.

Figure 9 shows how ice protection of the engine air intake fairings, spinners, and auxiliary air scoops for the jet pipe muff, and so on, is provided by the use of "Spraymat" electrical heaters, the power required being taken from a 40 kV.A. alternator on each accessory gearbox. The heater loadings are designed to give complete protection up to the specified B.C.A.R. and C.A.R. intermittent maximum ice conditions. Continuous heating is used on the engine intake fairing, but the spinner is cyclically heated in conjunction with the propeller blades; two propeller blades and one-half of the spinner area are switched on simultaneously, the cycle being 30 seconds on, 90 seconds off; or 60 seconds on, 180 seconds off, when light icing conditions only are encountered.

The alternator capacity is sufficient to cater for full protection on both engines in the event of an alternator failure.

A self-contained and self-emptying oil and fuel drains system is fitted, use being made of ram-air pressure to empty the collector tank when a moderate air speed has been achieved and the aircraft is well clear of the ground and airport buildings.

The drains involved are those from the fuel dump valve, metering unit and jet pipe. Sufficient capacity is provided within the container to enable four engine starts on the ground to be made, with the corresponding intermediate shut-down procedure, before it is necessary to drain the canister manually.

The ram-air connection is taken into the oil cooler air scoop and if the canister is overfilled during ground tests, the fact is readily apparent from fuel emerging from the ram-air connection at the duct entry.

The discharge pipe is taken aft to the rear of the nacelle, clear of the exhaust and all surrounding structure, thus avoiding airframe contamination.

The system has proved to be 100 per cent satisfactory and foolproof.

Engine controls within the power plant are of the conventional push-pull rod type in the compressor bay, but use is made of torque tubes through the turbine

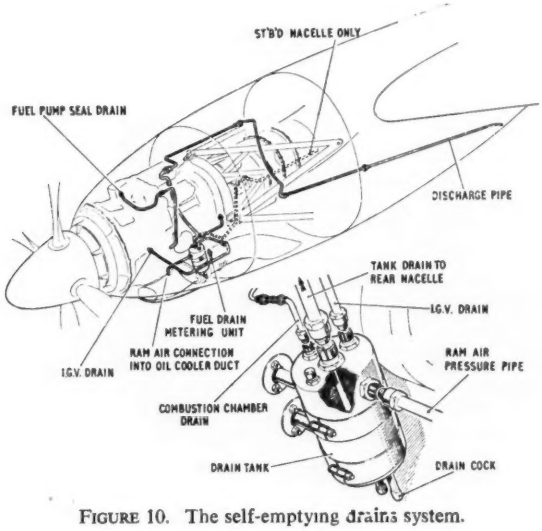


FIGURE 10. The self-emptying drains system.

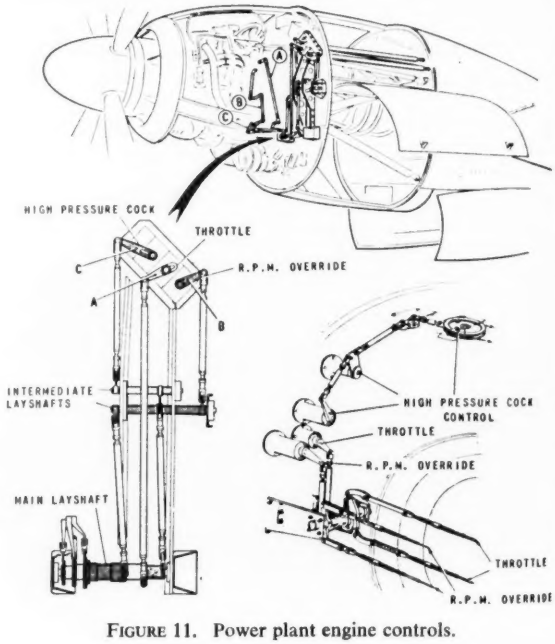


FIGURE 11. Power plant engine controls.

bay to reduce the effects of expansion due to the bay temperatures.

Universal joints are used at each end of the torque tubes to allow for general flexing and movement of the power plant. Control adjustments are provided for by means of slotted levers, radial vernier adjustment on the torque tubes and length adjustment on the push-pull rods.

## Aircraft Services

Existing aircraft services have been retained wherever possible. The main changes are as follows: —

All mechanical accessories were previously mounted on the back of the piston engines. With the turbo-propeller engine installed these same accessories are retained, but are now mounted on shaft-driven auxiliary gearboxes built on to the main fire wall. The accessories concerned are: —

(*i*) 9 kW. generator (port and starboard).
(*ii*) 3,000 p.s.i. hydraulic pump (port and starboard).
(*iii*) Cabin supercharger (starboard side only).
(*iv*) 40 kV.A. alternator (port and starboard). This is a new item, required primarily for the power plant de-icing system, and replaces a small alternator previously used for emergency instrument power only.

The introduction of the relatively large a.c. generating capacity has been made use of in other ways. An additional emergency d.c. supply has been arranged through a transformer/rectifier system, thus giving a third measure of protection on vital circuits.

The existing airframe of the Convair 340 aircraft, as with many present-day machines, uses a hot-gas anti-icing system. The hot air was previously obtained from muffs around the piston engine exhausts, and a muff has also been embodied on the turbine jet pipe to fulfil the same function. The cold and hot air passages are indicated in Fig. 12. The amount of heat extracted in this way is about 400,000 B.T.U.'s hour per engine; even so, this is only about two per cent of the available heat in the jet gases and thus does not affect engine performance in any way.

The degree of protection afforded by this system on the turbine-powered aircraft is somewhat superior to that of the original aircraft, which was only cleared by C.A.A. for operation in "light icing." It is hoped, ultimately, to obtain C.A.A. clearance to "maximum continuous" icing conditions, with the turbine-powered airframe.

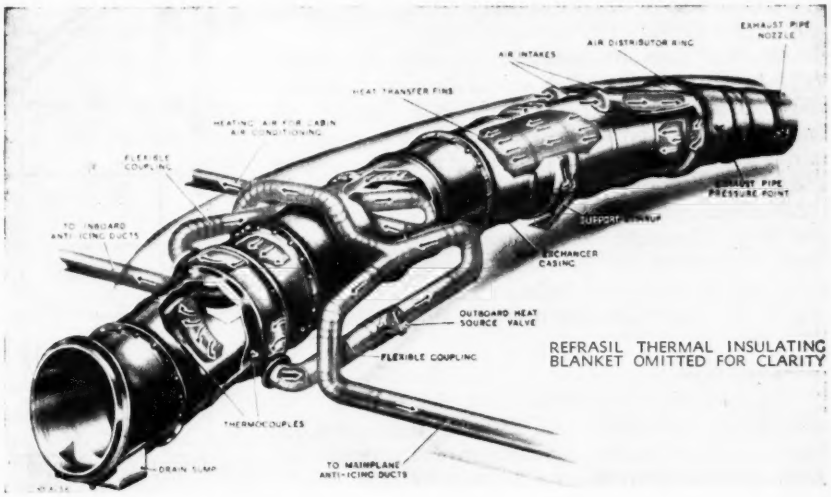


FIGURE 12. Exhaust pipe and hot air muff.

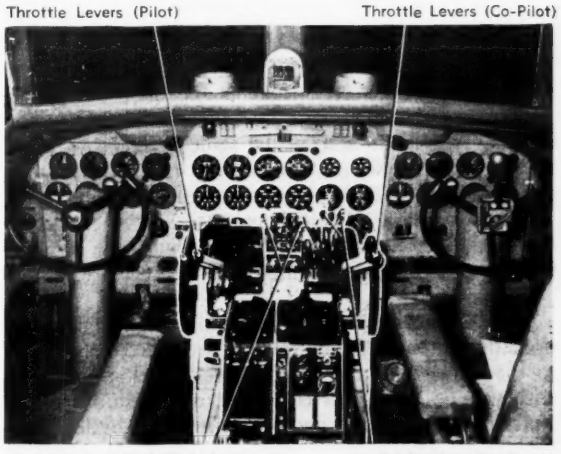


FIGURE 13. Cockpit layout and engine control pedestal.

Engine controls within the airframe are substantially unchanged, although the cockpit control pedestal has been re-arranged in a more logical manner to suit turbine engines.

Three controls are provided per engine: —

(*a*) Pilot's throttle lever.
(*b*) High-pressure fuel cock control, which embodies four separate functions, namely, "run," "dump," "positive coarse pitch" operation and "parking brake" operation.
(*c*) Engine r.p.m. over-ride, which enables engine speeds higher than interconnected values to be obtained without change in fuel flow.

No changes to the aircraft fuel tanks and pipes were found to be necessary as a result of using kerosine. The original positive-displacement booster pumps were replaced with British centrifugal types, mainly because of the superior vapour and air separating characteristics of the latter, particularly when used in conjunction with fuel recuperators for negative-*g* protection.

A small air/fuel heat exchanger to protect the low-pressure filter against ice formation operates from

compressor bleed air, and gives a 30°C temperature rise in the fuel.

Fuel flowmeters are fitted, combined "gallons gone" and "rate-of-flow" instruments being used. This facility has the advantage of enabling ground tuning checks to be made and, at the same time, provides an instrument upon which the pilot can assess quickly his available fuel reserves at any time during flight.

The hydraulic system was retained, to American standard, without any changes other than those involved in re-positioning the pumps on the new accessory gearboxes.

The cabin supercharger was modified to single speed operation instead of the previous two-speed system due to the reduced engine speed range, which is obtained with turbine engines as compared with piston engines. Full cabin conditioning and pressurising can be achieved, right down to engine idling speed.

Engine starting on the converted aircraft is a 120 Volt d.c. electrical system, with internal lightweight batteries giving four to six starts without recharging. These batteries are recharged in flight from the a.c. system through transformer/rectifiers. It is possible to fit an alternative air turbine system if required.

The starting cycle, to idling r.p.m., is designed to take about three-quarters of a minute per engine.

## Propellers

Four-blade, 13 ft. 6 in. diameter, propellers are fitted. These are of the constant-speed type and provision is made for two flight stops, known as the approach fine pitch and the take-off fine pitch stops.

The former is a mechanical abutment-type stop, designed to give the coarsest blade angle consistent with with good handling characteristics at low air speeds and minimum powers.

The take-off fine pitch stop is designed to reduce the drag of the windmilling engine in the event of engine failure on take-off and if the positive coarse pitch system has not operated. The stop is hydraulic in principle.

A back-up, electro-hydraulic stop is also provided, set at an angle just below the approach fine pitch stop, as a second line of defence should the blades for some reason pass through the mechanical stop.

All three stops are removable to enable the blades to go to the basic fine pitch setting for engine starting and "discing" braking. Reverse pitch, power on, braking can be provided if required, the propeller having been designed with this in mind. The engine and propeller control system has been designed throughout to meet existing A.R.B. and C.A.A. requirements and it is a principle of the design that at least two lines of defence be provided against any failure, electrical, mechanical or hydraulic. Thus a very high degree of safety of operation is attained.

## Performance

The calculated performance of the Convair 340 aircraft, when powered by two 3,500 h.p. turbo-propeller engines, is based on the requirements of American Civil Air Regulations, Part 4b.

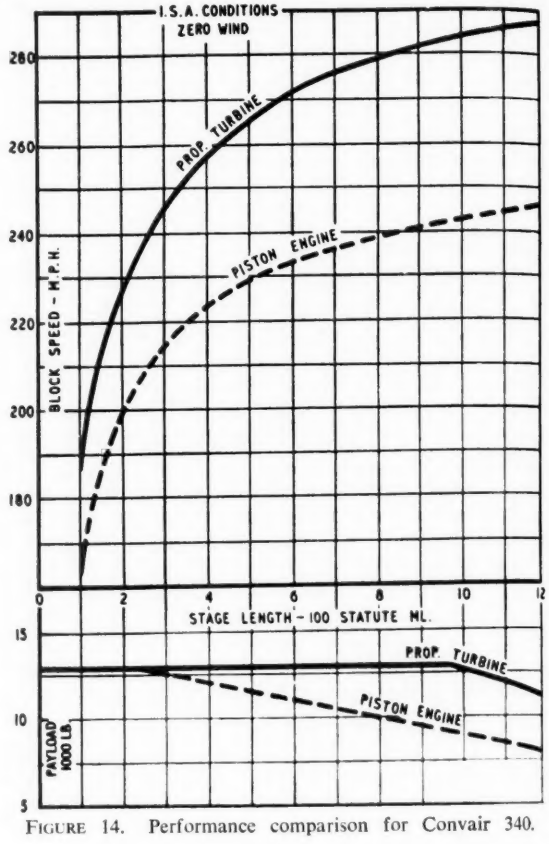


FIGURE 14. Performance comparison for Convair 340.

1. ———— Standard Convair 340—seat row 1 (forward)
2. – – – – – Standard Convair 340—seat row 11 (rear)
3. ———— Eland/Convair—seat row 1 (forward)
4. – – – – – Eland/Convair—seat row 11 (rear)

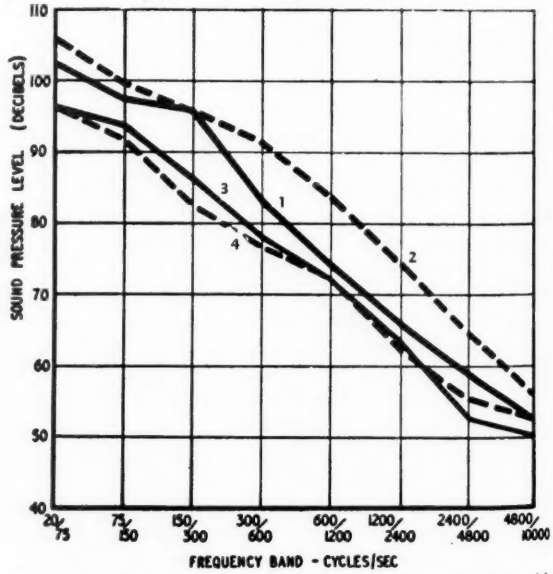


FIGURE 15. Comparison of noise levels in Convair 340 with piston and turbine engines.

Comparison of the new performance figures with those of the original aircraft show the following outstanding improvements: —

(*i*) The increase in take-off power is 2,200 h.p., achieved with a saving in weight of 500 lb.

(*ii*) At the maximum payload of 12,900 lb., the range has been increased from 250 miles to 970 miles, with normal fuel reserves.

(*iii*) For a typical 600 mile stage length the original piston-engined aircraft carried 11,250 lb. payload at a block speed of 233 m.p.h. When converted to the turbo-propeller installation, the capacity payload of 12,900 lb. can be carried at a block speed of 271 m.p.h. The payload range and block speed relationships are shown in Fig. 14.

(*iv*) Climbing time to cruise altitude is reduced by 60 per cent.

Advantage has been taken of the fact that, with these turbo-propeller engines, the maximum take-off weight under I.S.A. conditions can be increased to the airframe structural limit of 53,000 lb.

The installation of lighter engines with a greater "overhang" has inevitably affected the wing relief loads. These are partially offset by the increased fuel weight (due to use of kerosine), but the zero fuel weight has been decreased from 45,000 lb. to 44,000 lb. This has no appreciable effect on the operational characteristics.

With the turbo-propeller engine installed, a change of 1°C in ambient temperature results in a 0·72 per cent change in shaft horse power either up or down according to whether the temperature change is positive or negative. All quoted power ratings are related to I.S.A. conditions; hence, at I.S.A. plus 23°C (intercontinental maximum temperature) a 16·5 per cent fall in shaft horse power would be anticipated. The 150 h.p. reserve allowed by the torque-restriction setting does not entirely offset this loss, but does ensure that the performance at I.S.A. plus 23°C is equal to that to be expected from the equivalent piston engine.

The reduction in noise level in the aircraft cabin has been considerable, as was anticipated. The results are shown relevant to cabin seat rows one and 11, row one being the forward row. It can, therefore, be seen that a considerable reduction in the speech interference noise level has been achieved by installing turbo-propeller engines. The greatest contribution to this being a heavy reduction in the exhaust noise.

Performance assessments have also been made for the Douglas DC-6 and Lockheed Constellation 749.

It is seen that an increase in block speed of about 50 m.p.h. is achieved with the DC-6 conversion, but a small reduction in stage length with full payload also results. This is due to the fact that stress limitations on the existing airframe make it impossible to increase the all-up weight beyond that used at present, whereas the Convair 340 was able to operate effectively at the maximum design all-up weight when the greater power of the turbine engines was available.

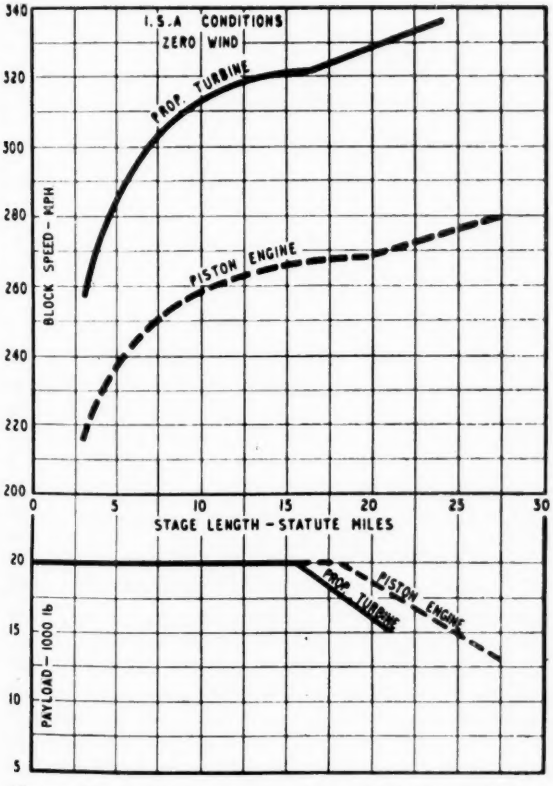


FIGURE 16. Performance comparison of Douglas DC-6B.

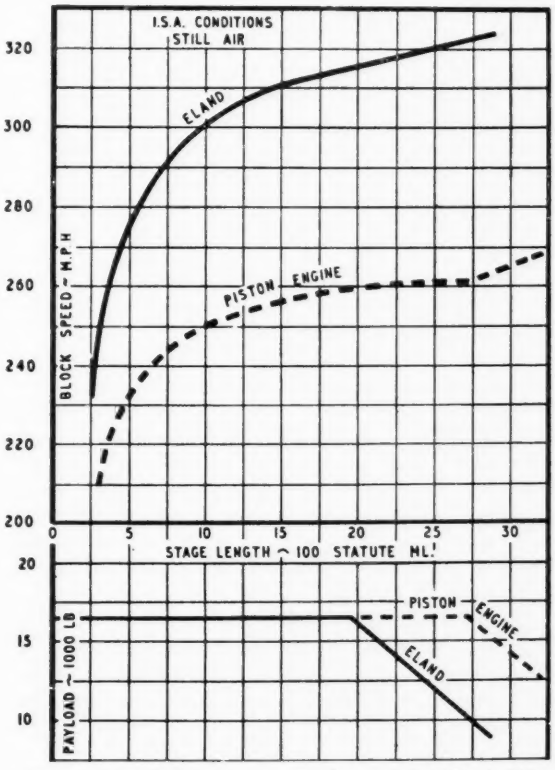


FIGURE 17. Performance comparison of Lockheed 749.

As with the DC-6, the Constellation 749 shows a 50 m.p.h. increase in block speed, but a similar reduction in stage length with full payload is observed, again due to weight limitations.

## Economics of Conversion

As previously stated, the conversion of an existing operating air liner can only be justified if greater profits can be earned after conversion than were possible before. The profit increase must be made up either from reduced operating cost or increased revenue earning capacity, or a combination of both. Again the Convair 340 is dealt with in detail, but overall results are given for the Douglas DC-6B and Lockheed Constellation 749.

As few of the aircraft under consideration were operating in Great Britain, it was thought that a more accurate comparison could be reached if the operating costs and revenue were based on operating data supplied by U.S.A. operators and the Civil Aeronautics Board, as published by the Air Transport Association of America under the title "Airline Financial and Operating Statistics." It was considered that this approach would be far more realistic than using one of the standard methods for the estimation of direct operating costs, since the standard methods (such as the S.B.A.C.) are useful for a broad comparison but are not suitable when considering the actual economics of internal routes. The figures that have been used are those reported by the larger U.S.A. operators for aircraft in domestic service. The figures have, for convenience, been converted into English currency.

The breakdown of the costs for the Convair 340 piston engined aircraft are shown in Fig. 18, together with the costs for the propeller-turbined version..

It must be appreciated that if the "costings" were done for operation by an operator in Great Britain, the whole level would be lower quantitatively on account of the difference between American and British labour costs, but this would not invalidate the comparison.

It will be evident that the majority of items in the breakdown of direct operating costs will be identical for the aircraft before and after conversion, as they are unaffected by the change in power plant. The items which are affected are as follows: —

In the "Flying Operations" section there is only one item, namely, the cost of pilots and co-pilots. It is the practice in the United States to pay pilots higher rates for higher cruising speed of the aircraft. Therefore, there is an increase from £10·75 per hour to £11·28 per hour. Another item in this section is the petrol tax. In the United States there is a tax on petrol but not on kerosine. There is, therefore, a total saving on the turbo-propeller version in this section amounting to £1·38 per flying hour.

Again, referring to Fig. 18, in the section "Direct Maintenance—Flight Equipment," the total costs of the turbo-propeller version is higher than the piston-engined version by £6·33 per flying hour. This is brought about by assuming that the cost of overhauling the turbo-propeller engines is considerably more than that of piston engines. The time will come when the overhaul

*Breakdown of Direct Operating Costs (Apart from Fuel and Oil)*

| *C.A.B. Account No.* | | *Piston Engine Version £ per hour* | *Prop. Turbine Version £ per hour* |
|---|---|---|---|
| 5,100 *Flying Operations* | | | |
| 23 | Pilots and Co-Pilots | 10·75 | 11·28 |
| 36 | Travel and incidental | 1·14 | 1·14 |
| 44 | Rentals | 0·22 | 0·22 |
| 53 | Other supplies | 0·41 | 0·41 |
| 55 | Insurance—Pub. Liab. Prop. Dam. | 1·78 | 1·78 |
| 69 | Taxes other than Payroll | 1·91 | — |
| 72 | Amort. other defr. charges | 0·47 | 0·47 |
| | | 16·68 | 15·30 |
| 5,200 *Direct Maintenance—Flight Equipment* | | | |
| 25 | Labour | 2·61 | 2·61 |
| 26 | Labour—Aircraft Eng. | 2·19 | 2·67 |
| 27 | Labour—Other | 0·66 | 0·66 |
| 46-1 | Materials—Aircraft | 0·81 | 0·81 |
| 46-2 | Outside Repairs—Aircraft | 0·77 | 0·77 |
| 47-1 | Materials—Aircraft Eng. | 5·71 | 12·14 |
| 47-2 | Outside Repairs—Aircraft Eng. | 0·58 | — |
| 48-1 | Materials—Other | 0·28 | 0·28 |
| 48-2 | Outside Repairs—Other | ·08 | ·08 |
| | | 13·69 | 20·02 |
| 5,900 *Depreciation—Flight Equipment* | | | |
| 75 | Aircraft | 7·90 | |
| 76 | Aircraft Engines | 1·41 | 11·11 |
| 77 | Other Flight Equipment | 1·18 | |
| | | 11·09 | 11·11 |
| *Summary* | | | |
| Total | Flying Operations (Other than fuel and oil) | 16·68 | 15·30 |
| Total | Direct Maintenance Flight Equipment | 13·69 | 20·02 |
| Total | Depreciation — Flight Equipment | 11·09 | 11·11 |
| | | 41·46 | 46·43 |

FIGURE 18. Comparison of Convair 340 direct operating cost breakdown.

cost of gas turbine engines will be less than that of the piston engines, but as this may not be achieved for some years, I felt it advisable to take a cautious figure owing to the lack of experience and knowledge of this particular item at the present time, but it is based on figures which engine manufacturers are prepared to guarantee. I appreciate that this figure can be criticised as there is already evidence in practice that the maintenance costs of turbo-propeller engines are lower, but there is insufficient "mass" experience of this new type of engine to be categorical; but, in spite of taking this conservative view, final results show that it pays handsomely to convert aircraft to turbine engines.

Dealing with the section on depreciation, the depreciation cost for the turbo-propeller air liner has been based on the assumption that the aircraft is converted five years after the initial purchase. At this time, the aircraft has a certain book value and to this value is added the total cost of conversion, including engines,

propellers and spares. A new depreciation rate is then determined on a further seven years of life at the same annual utilisation of the aircraft as previously achieved. Fig. 19 shows a typical calculation.

The period of five years elapsing before conversion was chosen as being representative, assuming the conversion would be made by early 1958. However, it is interesting to note the effect of actual conversion time on the direct operating cost. The lower curve of Fig. 20 shows the variation of depreciation rate with time of conversion and the upper curve shows the effect of this on the direct operating cost. The propeller-turbine cost has been plotted as a percentage of the piston engine over stage distances of 200/800 miles. This ratio does not vary appreciably with stage distance. Here again a further seven years of operation after date of conversion and the same annual utilisation have been assumed. Only if the revenue earning power of the turbo-propeller aircraft is higher would it be justifiable to convert earlier than, say, after two years of operation from the date it first went into service.

The costs which have been dealt with are called "Fixed Costs" which are independent of stage length of flight and the "Variable Costs" must now be calculated to arrive at the total direct operating cost.

Figure 21 shows details of the "Variable Costs," together with the total direct operating cost and shows costs in "Pence per Aircraft Mile" for five examples of stage lengths from 100-800 miles. In all these calculations the cost of petrol and oil for the piston engines has been taken at 15·33 pence (17·9 cents) per U.S. gallon and 38·6 pence (45 cents) per U.S. gallon respectively. For Aviation Kerosine and Lubricating Oil for the turbo-propeller, the costs are 8·56 pence (10 cents) and 50/- ($7) per U.S. gallon respectively. It will be found that this relationship in the costs of the fuel does not vary much throughout the World. It will

| | *Piston Engine Version* | *Prop. Turbine Version* |
|---|---|---|
| First cost—£ ... ... ... | 215,700 | |
| Add 20 per cent spares ... | 43,140 | |
| | 258,840 | |
| Less 10 per cent residual value | 25,880 | |
| | 232,960 | |
| Depreciation per annum (7 year period) ... ... | 33,280 | |
| Depreciation per hour (3,000 hours utilisation) ... | 11·09 | |
| Written off value after 5 years | 166,400 | |
| Book value after 5 years ... | 92,440 | 92,440 |
| Add conversion cost ... ... | | 125,770 |
| Add 50 per cent spare engines | | 41,000 |
| | | 259,210 |
| Less 10 per cent residual value | | 25,920 |
| | | 233,290 |
| Depreciation per annum (7 year period) ... ... | | 33,327 |
| Depreciation per hour (3,000 hours utilisation) ... | | 11·11 |

FIGURE 19. Depreciation rates for Convair 340.

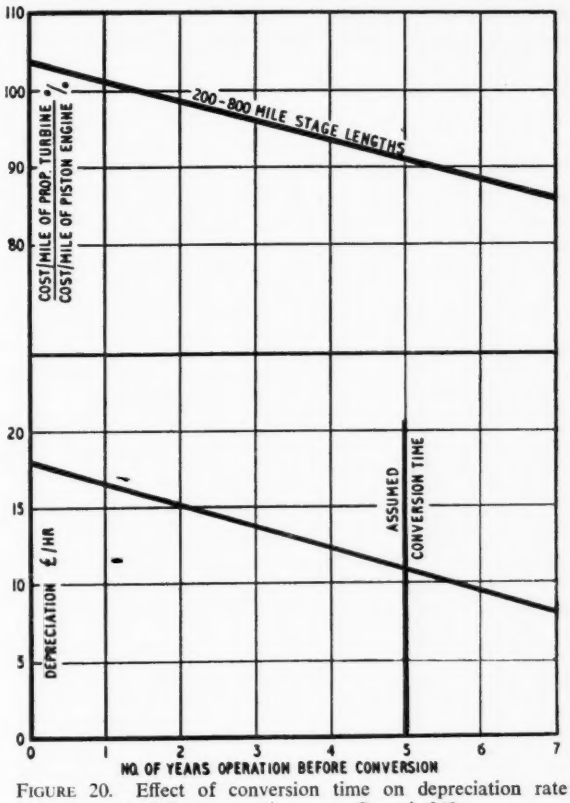


FIGURE 20. Effect of conversion time on depreciation rate and direct operating cost—Convair 340.

be noted that in comparing the two aircraft, in spite of the direct operating cost of the turbo-propeller version being higher than the piston engine version, the "Variable Costs" expressed in cost per mile are approximately 10 per cent less. This is due to the block speed of the aircraft, *i.e.* time from take-off to landing, being considerably higher and the machine is able to do more trips in a year and so spread the direct cost over a greater number of miles flown.

| | *Direct Cost £/hr* | *Oil Cost £/hr* | *Fuel Cost £/hr* | *Total Cost £/hr* | *Block Speed mph* | *Cost/ Mile Pence* |
|---|---|---|---|---|---|---|
| 100 *Mile* | | | | | | |
| Piston Engine | 41·46 | ·51 | 15·14 | 57·11 | 167 | 82·07 |
| Prop. Turbine | 46·43 | ·62 | 11·80 | 58·85 | 190 | 74·34 |
| 200 *Mile* | | | | | | |
| Piston Engine | 41·46 | ·51 | 13·61 | 55·58 | 200 | 66·7 |
| Prop. Turbine | 46·43 | ·62 | 11·05 | 58·10 | 228 | 61·15 |
| 300 *Mile* | | | | | | |
| Piston Engine | 41·46 | ·51 | 12·93 | 54·90 | 215 | 61·27 |
| Prop. Turbine | 46·43 | ·62 | 10·71 | 57·76 | 246 | 56·35 |
| 500 *Mile* | | | | | | |
| Piston Engine | 41·46 | ·51 | 12·32 | 54·29 | 229 | 56·9 |
| Prop. Turbine | 46·43 | ·62 | 10·25 | 57·30 | 266 | 51·70 |
| 800 *Mile* | | | | | | |
| Piston Engine | 41·46 | ·51 | 12·07 | 54·04 | 239 | 54·26 |
| Prop. Turbine | 46·43 | ·62 | 10·18 | 57·23 | 279 | 49·22 |

FIGURE 21. Comparative summary of direct operating costs for Convair 340.

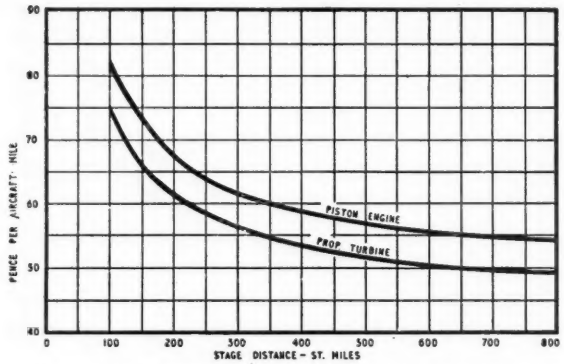


FIGURE 22. Comparison of direct operating costs for Convair 340.

| | 200 *St. Mile Stage Length* | |
|---|---|---|
| | *Piston engine* | *Prop. turbine engine* |
| Total direct operating cost £/hr. | 55·58 | 58·1 |
| Total indirect operating cost £/hr. ... ... ... | 55·58 | 58·1 |
| Total operating cost £/hr. ... | 111·16 | 116·2 |
| Block speed m.p.h. ... ... | 200 | 228 |
| Revenue per hour (at 5 pence per pass. mile and 65 per cent pass. load factor) £/hr. | 119 | 136 |
| Therefore profit per hour £/hr. | 7·84 | 19·8 |
| Profit per annum (3,000 hrs. utilisation) £/hr. ... | 23·520 | 59·400 |

FIGURE 23. Comparison of earning capacity for Convair 340.

It will be appreciated that the flying cost that matters is the cost per mile and not the cost per hour, because the cost must bear a direct relationship to the revenue-earning capacity of the aircraft which is determined by the number of passengers carried in one year. Fig. 22 shows graphically the direct operating costs for both machines for different stage lengths.

In determining the effect of the conversion on the profit-earning capacity of the two types of air liners, it is necessary to add the "Indirect Costs" to the "Direct Costs" and so obtain the complete cost of the airlines' operation. These Direct Costs include such items as General Administration, handling of all traffic on the ground, selling organisations' expenses, and so on. It has been found that such costs equal approximately the Direct Costs and, to arrive at the profit earned, I have assumed this. Fig. 23 shows a computation of profit earned per annum on the two types of Convair aircraft when operating on a 200 mile stage length. The revenue has been taken at five pence per passenger mile based on an average load factor of 65 per cent. A utilisation of 3,000 hours per year has been assumed for each aircraft and it can be seen that the turbo-propeller machine earns £35,880 per year more profit than the piston-engined aircraft.

It can be calculated from the information contained in Fig. 23 that this increase in profit earned per annum by the turbo-propeller version, increases with increase of stage length and, at stage lengths of 800 miles, the profit difference per air liner per annum is £51,780, *i.e.* the extra profit earned on a fleet of 20 machines would be over £1,000,000 per year.

It is also probably true that where there are piston-engined air liners and turbo-propeller ones operating on the same route, the more modern machine with its greater comfort and less time spent on the journey will attract a bigger proportion of the traffic. If so, then the revenue on the turbo-propeller air liner will be increased above the 65 per cent load factor taken in the profit calculations and that for the piston-engined air liner would be correspondingly reduced, so that the profit differential in favour of the turbo-propeller liner will be greater still.

Assessment of direct operating costs has also been made for the Douglas DC-6B and Lockheed Constellation 749 aircraft, the results of which are shown in Figs. 24 and 25. Here too it will be noted that similar reductions in cost per aircraft mile can be achieved.

This then is the case for the conversion of existing piston-engined aircraft to propeller-turbine power. By this means airline operators will be able to offer to the public this much advanced form of air transport sooner than would otherwise be possible, by the natural replacement of piston-engined air liners by new turbo-propeller or jet engines.

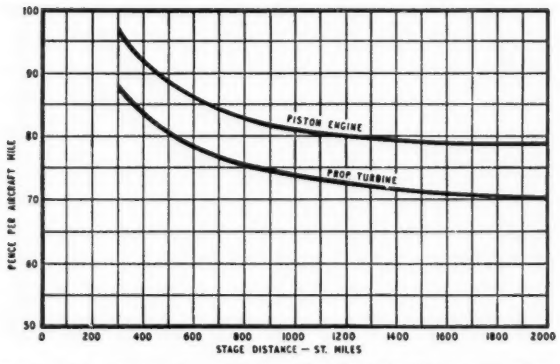


FIGURE 24. Comparison of Douglas DC-6B direct operating costs.

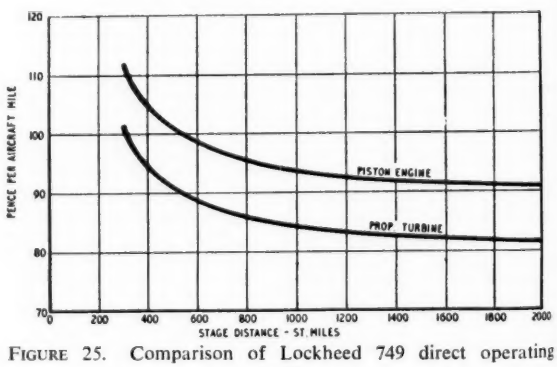


FIGURE 25. Comparison of Lockheed 749 direct operating costs.

# The Effects of Kinetic Heating on Aircraft Structures*

by

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## 1. Introduction

Kinetic heating, which is associated with supersonic flight, is probably the greatest challenge which has confronted the structural engineer, since the design and analysis of affected structures requires a radical extension of current practice. For example, the laws of heat transfer and the effects of high temperature on structural materials become important factors in structural analysis. To obtain a clear appreciation of the situation it is therefore necessary for the designers of the structure to become familiar with these factors, together with certain aspects of boundary layer theory.

It is unfortunately impossible to cover adequately the many aspects of the problem in one paper, and it has been necessary to restrict the discussion to metallic materials, and to flight conditions such that the maximum structure temperature is approximately 500°C.

The first parts of the paper consider briefly the generation of heat within the boundary layer, together with the fundamentals of heat transfer to and within the structure, and the effects of temperature on material properties. This leads naturally to discussion of the effects of temperature and temperature gradient upon the structure. Some methods for alleviating the problems introduced are reviewed and finally design requirements are considered. It is regretted that space will not allow a review of the many problems involved with the representative testing of the structures, or of those arising during manufacture.

NOTATION

| | |
|---|---|
| $A$ | cross-sectional area |
| $c$ | specific heat |
| $E$ | modulus of elasticity |
| $E_T$ | tangent modulus |
| $h$ | boundary layer heat transfer coefficient |
| $k$ | coefficient of thermal conductivity |
| $L$ | effective length of column or panel |
| $M$ | Mach number (local) |
| $P$ | panel buckling load per unit width |
| $P_{cr}$ | column buckling load |
| $q$ | heat flux due to forced convection |
| $q_R$ | heat flux due to radiation |
| $r$ | temperature recovery factor |
| $t$ | time |
| $T$ | temperature |
| $T_{amb}$ | air temperature, ambient |
| $T_t$ | total, or stagnation, temperature |
| $T_w$ | skin temperature |
| $T_{wo}$ | adiabatic wall temperature |
| $\alpha$ | coefficient of thermal expansion |
| $\gamma$ | ratio of the specific heats for air |
| $\varepsilon$ | surface emissivity |
| $\rho$ | density |
| $\sigma$ | stress |
| $\sigma_{cr}$ | buckling stress. |

*Units*

The units used in any part of the analysis must of course be consistent and, in general, it is found that, for the length measurements, it is more convenient to use the foot in the heat transfer calculations, and the inch in the stress analysis. It is common practice in the United Kingdom to use the Centigrade temperature scale in heat transfer calculations, and it is important to note that the absolute scale (Kelvin) must be used in equations (1), (2) and (4).

## 2. The Heat Transfer Problem

The first consideration in the analysis of the structural problem is the determination of the environment, and the consequent temperature distribution within the structure. This involves an examination of the aerodynamic generation of heat and its subsequent transfer by convection, radiation and conduction; the fundamental principles of which are reviewed in this section.

### 2.1. KINETIC HEATING

The air adjacent to the surface of an aircraft (that is, within the boundary layer) is suddenly accelerated to the aircraft velocity. At the forward extremities this is brought about by normal forces with negligible heat transfer, that is, the process is an isentropic compression. The work done by the accelerating forces increases the energy of the air and appears as an increase in temperature to the total or stagnation temperature, which is given by

$$T_t = \left[1 + \left(\frac{\gamma - 1}{2}\right) M^2\right] T_{amb} \quad . \quad . \quad (1)$$

Aft of the stagnation points the air is accelerated by viscous forces under conditions in which heat transfer may occur within the boundary layer. Under these non-isentropic conditions the total temperature will not be achieved and, in its place, the reference temperature is taken to be the "adiabatic wall" or zero heat transfer temperature. This is the equilibrium temperature which would exist in an ideal insulated surface

*A lecture given to the Graduates' and Students' Section of the Society on 23rd November 1956.

having zero radiative heat exchange, and is given by

$$T_{wo}=\left[1+r\left(\frac{\gamma-1}{2}\right)M^2\right]T_{amb} \quad . \quad . \quad (2)$$

In this expression $r$ is called the temperature recovery factor and is approximately 0·85 in the laminar boundary layer, and 0·88-0·9 in the turbulent regions. It does not appear to be strongly dependent upon either Mach number or Reynolds number.

Using the adiabatic wall temperature as a measure of the intensity of the heat source, Newton's Law of Cooling may be written as

$$q=h(T_{wo}-T_w) \quad . \quad . \quad . \quad (3)$$

The heat transfer coefficient is related to the skin friction coefficient by the Reynolds Analogy, and is a function of ambient atmospheric properties, aircraft velocity and wall temperature. The coefficients are determined on a semi-empirical basis from test results and there are many reports and papers available giving further details (*see* for example Refs. 1 and 2).

### 2.2. RADIATION

In addition to heat transfer by forced convection there are further exchanges at the surfaces by radiation. The main effect is the heat lost from the skin according to the Stefan-Boltzman Law

$$q_R=2{\cdot}78\,\varepsilon\left(\frac{T}{1{,}000}\right)^4 \text{ C.H.U./ft.}^2\text{ sec.} \quad . \quad (4)$$

The effect is important where the skin temperature is high and the heat transfer coefficient is small, as in high speed, high altitude flight. The resultant equilibrium temperature of an insulated skin can be found by equating the heat received from the boundary layer to that lost by radiation, thus

$$h(T_{wo}-T_w)=2{\cdot}78\,\varepsilon\left(\frac{T_w}{1{,}000}\right)^4 . \quad . \quad (5)$$

The solution of this equation for certain conditions is shown in Fig. 1; the high values of the equilibrium temperature for Mach numbers greater than 2·5 to 3, and the effects of the loss due to radiation are immediately apparent by comparing the adiabatic wall temperatures ($\varepsilon=0$) with those existing in the presence of radiation. A constant specific heat was assumed in the calculations and this leads to a slight over-estimation of the higher temperatures.

Under certain conditions the skin will also receive heat by radiation from the sun, the maximum value of the flux being $7\times10^{-2}$ C.H.U./ft.$^2$ sec. and is, in general, small compared with the convective heat transfer. Since it can be reduced by surface finish it is usually neglected, as is the heat received by radiation from the surrounding air.

A further source of radiation which may affect the design of military aircraft is that due to atomic explosions. In addition, the structure will receive heat from the hot components contained by it, principally the power plant. The effects are local and can be controlled to some extent by surface finish and the use of insulation.

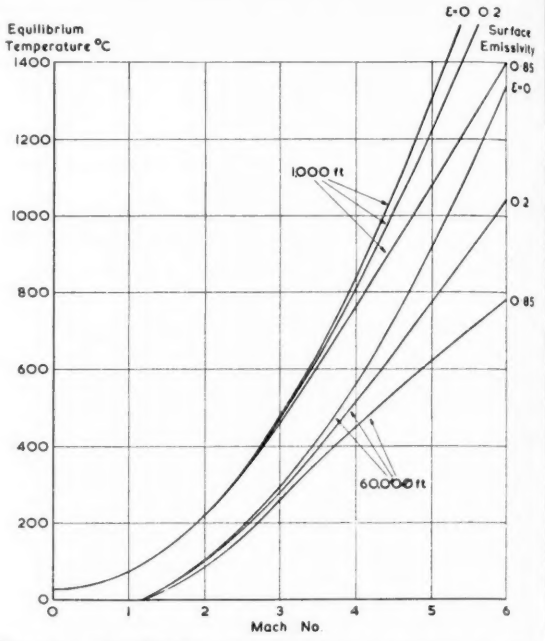



FIGURE 1. Equilibrium temperatures for a turbulent boundary layer ($l=5$ ft.) (I.C.A.N. standard atmosphere).

### 2.3. HEAT TRANSFER WITHIN THE STRUCTURE

One of the major problems in thermal stress analysis is the determination of the transient temperature distribution within the structure. The principal mode of the internal heat transfer is conduction, assisted by radiation at high skin temperatures.

The general equation governing heat transfer by conduction is

$$\nabla^2T=\frac{c\rho}{k}\frac{\partial T}{\partial t} \quad . \quad . \quad . \quad (6)$$

The solution of this equation is required under boundary conditions which, in general, vary with time at the outer surface. The problems which arise in practice are usually not those which are amenable to exact solution and approximate methods of analysis are used. In many cases the problem can be simplified into a quasi-one dimensional type and a step-by-step solution of the finite difference equations obtained. This work may be facilitated by the use of an analogue representation of the heat flow problem using an electrical circuit. A machine based on this analogy has been built by the Company* and the temperature distributions in the examples given in this paper have been obtained by its use. In many cases a reliable estimate of skin temperature can be obtained by neglecting heat flow within the structure and equating the net heat flow into a skin element to the rate of increase in the total heat of the skin.[3] This treatment fails in the regions of large temperature gradients along the structure, *e.g.* near the leading and trailing edges and in the vicinity

*This machine is based on a design given by H. G. R. Robinson of the Royal Aircraft Establishment in an unpublished R.A.E. report.

of spar webs. In these cases it is necessary to include terms describing conduction along the skin and it is, in general, necessary to resort to the step-by-step solutions discussed in several papers by Schuh[4], or to an analogue representation of the problem.

In cases where the skin temperature is greater than about 400°C it may be necessary to make an allowance for radiation within the structure, these effects have been discussed by Hoff[5], and Bisplinghoff[6] in recent papers. The importance of radiation depends upon the ratio of surface emissivity to thermal conductivity and is more important for titanium and stainless steel structures than for those using aluminium alloys.

A further mode of internal heat transfer is natural convection but this is generally negligible.

In actual structures the joints form discontinuities in the heat flow path, the magnitude of the associated resistance depends upon many factors and test work will usually be required so that due allowances may be made during the analysis.

## 3. The Effect of High Temperatures on Material Properties

The effects of kinetic heating on the structure may be divided conveniently into three classes, namely,

(*a*) The effect of high temperatures on material properties.

(*b*) The effect of high temperatures on structural elements.

(*c*) The effect of high temperature gradients within the structure.

To demonstrate the effects on material properties the deterioration of a few possible structural materials at elevated temperatures will be considered. (The materials discussed were chosen mainly because sufficient data for a reasonable evaluation was readily available, *see* References 7 to 17 inclusive).

### 3.1. MATERIAL PROPERTIES UNDER HIGH RATES OF LOADING

In this section it is assumed that the rates of loading are sufficient to eliminate any creep effects but not to introduce dynamic effects. The length of the exposure time to elevated temperature which has to be considered depends upon the role of the airframe and will range from seconds for a missile structure, to hundreds of hours for an aircraft. In some cases it may be necessary to consider the mechanical properties at room temperature levels after exposure to high temperatures, *e.g.* for structure forming undercarriage attachment.

#### 3.1.1. *Short Exposure*

Any finite exposure time at a sufficiently high temperature will cause a marked reduction in mechanical properties and, as an example, the ultimate tensile strength and 0·2 per cent proof stress existing in tests made at the given temperature following one hour's exposure to the temperature are given in Fig. 2 for a few representative materials. The stress ordinate is given on a specific basis to facilitate a comparison of relative efficiency.

For ultimate tensile strength considerations the marked superiority of the titanium alloy C-110M below 200°C is evident and this is only slightly less efficient than the precipitation hardening stainless steel (Armco 17-7PH) up to temperatures of about 470°C. The rapid deterioration of the high strength aluminium alloy L65/L71 is most marked and becomes serious above about 150°C.

The position changes when the proof stress is considered and stainless steel has the highest values over the majority of the temperature range considered. The deterioration of aluminium alloys above 150°C is again evident, but the most striking effect is the rapid deterioration of the titanium alloy between room temperature and 200°C. The relatively stable behaviour of the (73 per cent) Nickel-Chromium alloy Inconel X is also noticeable; this alloy is equivalent to the British Nimonic alloys and should not be confused with the British Inconel.

In both sets of curves one of the most important features is the decrease in maximum possible efficiency with temperature.

#### 3.1.2. *Extended Exposure*

The most pronounced effects of exposure time occur for aluminium alloys at temperatures greater than 100°C, this being due to over precipitation of the hardening constituent. In certain alloys, such as L.72 and the American alloy 2024, exposure times up to 100 hours at approximately 150°C may lead to an increase in strength due to

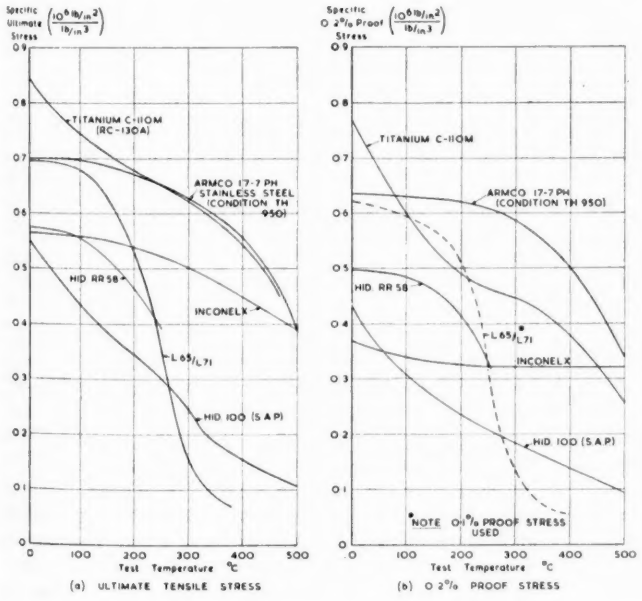



FIGURE 2. Mechanical properties after one hour soak at test temperature.

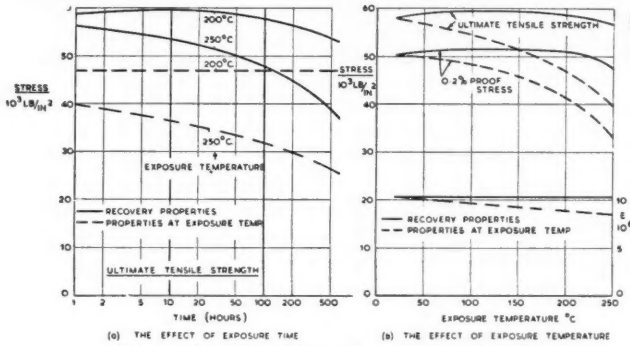



FIGURE 3. The effects of exposure temperature and time on Hiduminium RR58

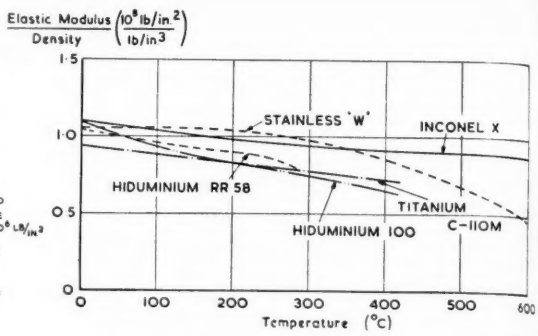



FIGURE 4. The effect of temperature on specific stiffness.

precipitation hardening, but further exposure leads to over-precipitation and a decrease in properties. The sintered aluminium powder (Hiduminium 100) retains its properties up to 100 hour exposures, but shows some loss in ductility above 200°C.

The effect on the other materials discussed is negligible, although under extreme conditions titanium alloys show some loss in ductility.

3.1.3. *Recovery Properties*

The reduction in the strength of the aluminium alloys at elevated temperatures is largely due to a modification of the structure brought about by precipitation. Since the process is irreversible there is also a reduction in strength when the material is tested at room temperature after exposure; these properties are the recovery properties. The curves of Fig. 3 show the effect of exposure time and temperature upon the recovery properties, and a comparison with properties at the elevated temperatures for the High Duty Alloys Ltd. material, Hiduminium RR58[7]. The effect varies with the actual alloy considered and, as stated previously, under certain conditions, some show an increase in strength; it should be noted that RR58 is less sensitive to prolonged exposure than many other aluminium alloys and for this reason is becoming popular for moderately hot structures. For this material it will be noted that the effect on recovery properties does not become marked until the exposure temperature is greater than 200°C and is then appreciably affected by exposure time. The modulus of elasticity, however, exhibits almost complete recovery.

These effects are much less pronounced in other materials and in many cases are negligible; this includes the sintered aluminium powder for which the only factor seriously affected is the limit of proportionality.

3.1.4. *Stiffness*

The effect of temperature on stiffness is illustrated by the curves of Fig. 4 in which the variation of the specific elastic modulus is shown for the materials previously considered. These curves give a guide to material selection where stiffness is the design parameter. At room temperature there is little to choose between them. The titanium alloy shows a steady decrease (as does Hiduminium RR58) and does not become superior to RR58 until approximately 280°C, at this point RR58 is deteriorating rapidly. The alloy Inconel X shows a very slight steady decrease in stiffness up to at least 650°C and becomes the most efficient material at temperatures above 350°C, displacing the precipitation hardening stainless steel which then starts to deteriorate rapidly.

The effects on the modulus of rigidity will be similar, but slightly more marked, as there is a tendency for Poisson's ratio to increase with temperature.

3.2. OTHER MECHANICAL AND PHYSICAL PROPERTIES

There is incomplete data relating to the remaining mechanical properties apart from creep. However, it is worthwhile noting that there is reduction in fatigue strength with increase in temperature, although the effect is not so marked as for the ultimate and proof strengths.

It is also important to note that the physical properties which affect the present problem, namely thermal conductivity, specific heat and coefficient of expansion, are also functions of temperature which increases the difficulty of exact thermal stress analysis.

3.3. CREEP

It will be seen later that creep has an important effect on the behaviour of a hot structure. The significance will depend upon the role of the airframe and may be negligible for short exposure to temperature, for example missile structures and perhaps even in some interceptor aircraft. To illustrate the differences between material types Fig. 5 shows the effect of temperature on the stress to produce 0·1 per cent deformation in 1,000 hours, and that to cause rupture in 100 hours and 1,000 hours. These curves are again given on a specific basis to facilitate a comparison of material efficiency. The effects on aluminium alloy are most marked even for relatively short lifetimes. There is, at present, little data available relating to the creep of titanium alloys; however, creep in this case occurs also at room temperature. Rem-Cru state that near to room temperature and above 370°C, creep will occur in titanium at stresses below the yield point but between this range stresses considerably greater than the yield value are

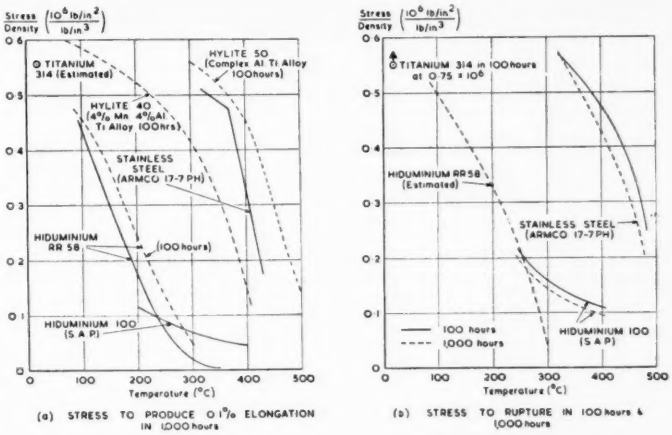

Stress / Density (10⁶ lb/in² / lb/in³)
TITANIUM 314 (Estimated)
HYLITE 50 (Complex Al Ti Alloy 100hours)
HYLITE 40 (4% Mn 4% Al Ti Alloy 100hrs)
STAINLESS STEEL (ARMCO 17-7 PH)
HIDUMINIUM RR 58
(100 hours)
HIDUMINIUM 100 (S A P)
Temperature (°C)
(a) STRESS TO PRODUCE 0·1% ELONGATION IN 1,000 hours
TITANIUM 314 in 100hours at 0·75 × 10⁶
HIDUMINIUM RR58 (Estimated)
STAINLESS STEEL (ARMCO 17-7PH)
100 hours
1,000 hours
HIDUMINIUM 100 (S A P)
(b) STRESS TO RUPTURE IN 100 hours & 1,000 hours

FIGURE 5. Specific creep strengths.

required to produce appreciable creep. Creep stresses required to produce 0·1 per cent deformation in 100 hours for two types of titanium alloy are also given.

The precipitation hardening stainless steels allow a high proportion of the short time properties to be used up to temperatures of almost 400°C, and for lifetimes of about 1,000 hours. It will be noted that, in all cases, creep strengths show a marked dependence on temperature.

The foregoing data relates to conditions of steady creep. There is, however, little evidence available relating to alternating or cyclic loading except for the following indications[31]:

(*a*) Small alternating stresses superimposed upon a mean stress have only a slight effect on creep.

(*b*) To a first approximation the total creep under cyclic loading is due to the net time spent at each stress level.

## 4. The Effects of High Temperature on Structural Elements

The reduction of mechanical properties due to high temperature naturally influences the behaviour of structural elements operating under conditions of uniform elevated temperature. The deterioration of some typical structural elements is considered in this section, with the restriction that thermal and temperature stresses do not exist (*i.e.* the elements are homogeneous and free from temperature gradients.)

The efficiency of the tensile elements can be assessed directly from the curves of specific ultimate tensile strength previously discussed and, where applicable, creep and stress rupture data.

The stresses arising due to the connection of materials having different coefficients of expansion will not be considered since the effect is too well known to need further comment.

The loss in stiffness of the structural material will lead to a reduction in flexural and torsional stiffness as the temperature is increased; this must be accounted for in aeroelastic calculations. The effect is less than 10 per cent below 150°C for aluminium alloys and 320°C for stainless steel; where stiffness is the prime consideration alloys of the Inconel X type (Nimonics) are the correct choice above 350°C.

The remaining major effects are those affecting the stability of bars and panels, and the influence of creep on stability and stress distribution. The first consideration is given to elements subjected to high rates of loading following an exposure to elevated temperatures, so that any creep effects are negligible.

### 4.1. STRUTS AND INTEGRALLY STIFFENED SKINS

The efficiency of columns has been examined by Heimerl and Barrett[18] with the restriction that only the flexural mode of failure exists. The influence of temperature on the local modes can be determined by studying plate buckling. Their derivation of a suitable structural index uses the Shanley hypothesis for inelastic buckling, and is obtained by including a shape parameter, $f$, (equal to the square of the radius of gyration divided by the cross-sectional area) and the fixation constant, $c$.

$$\text{Thus:} \quad \frac{P_{cr} c f}{L^2} = \frac{\sigma_{cr}^2}{\pi^2 E_T} . \quad . \quad . \quad . \quad (7)$$

Using the variation of the tangent modulus with stress for a given temperature and material, a curve giving the variation of specific buckling stress against the structural index

$$\frac{P_{cr}\, cf}{L^2}$$

is obtained, thus enabling the relative efficiency of materials to be examined.

The design of optimum integrally stiffened panels at room temperature has been considered by Catchpole[19]; he shows that for optimum proportions the buckling stress is given by

$$\sigma_{cr} = 0{\cdot}81 \sqrt{\frac{PE_T}{L}} . \quad . \quad . \quad (8)$$

($P$ being the load per inch of width).

A similar expression will also apply to other types of optimum skin-stringer panels provided the appropriate constant is used, *see* Farrar[20].

If it is assumed that this result will also apply at high temperature, then by solving for the structural index we obtain:

$$\frac{P}{L} = 1{\cdot}525 \frac{\sigma_{cr}^2}{E_T} . \quad . \quad . \quad (9)$$

Thus by suitably scaling the abscissa the column curves will describe the efficiency of integral panels. The results of a series of calculations of this type for the materials previously considered are shown in Fig. 6;

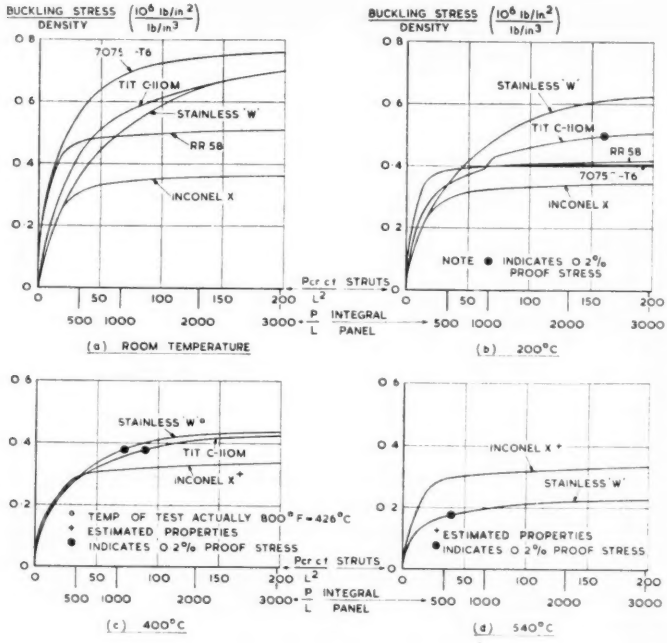



FIGURE 6. Relative efficiencies of materials for columns and integral panels following one hour exposure to temperature.

curves for the aluminium-zinc alloy 7075-T6[17] have been included as this is representative of the high strength aluminium alloys currently used. Stainless W has been used to represent the precipitation hardening stainless steels as tangent modulus data was readily available; this steel is superior to Armco 17-7PH at temperatures below 400°C unless the latter can be used in the cold rolled and hardened form CH900.

The rapid deterioration of the aluminium alloys represented by 7075-T6 is apparent and this becomes less efficient than RR58 at high loadings at 200°C. However, at 200°C there will still be a large number of cases where aluminium alloys provide the most efficient integral skin panels. It is also apparent that for moderate and high loadings titanium alloys fall below Stainless W at 200°C but are almost as efficient at 400°C. The high temperature alloy Inconel X shows little change in efficiency with increasing temperature and assumes the leading position at about 500°C.

The most noticeable feature of this family of curves is the general lowering of efficiency with increasing temperature. The effect of this may be seen by considering the weights of integral panels for given values of structural index at high temperature relative to those at room temperature, assuming that the most efficient material is used in each case; a few examples are given in Table I.

TABLE I

| *Structural Index* | *Weight at temperature / Weight at room temperature* | |
|---|---|---|
| $P/L$ | 200°C | 400°C |
| 250 | 1·14 | 1·85 |
| 1,000 | 1·45 | 1·83 |
| 3,000 | 1·22 | 1·74 |

At 200°C the effect of temperature is most marked for values of $P/L$ near to 1,000 where there is a 45 per cent increase in weight. At 400°C the effect is most marked at low loadings but always causes more than a 74 per cent increase in weight. It seems appropriate to emphasise at this point that these results apply only to elements designed for compressive loading, and the weight increases are more severe than those for the total structure weight.

### 4.2. PLATES SUBJECT TO COMPRESSION

A similar analysis of the relative efficiency for plates designed to withstand buckling has also been given by Heimerl and Barrett[18]. However, Mathauser and Deveikis[21] have derived an empirical relationship giving the failing stress of plates in compression which takes account of post-buckling characteristics and correlates well with stiffened panel experiments. The result has been shown to be applicable to all materials at room temperature, to aluminium alloys at elevated temperatures, and can be extended to cover creep strengths. Assuming that the relationship derived also applies to other materials at elevated temperatures, which does not appear to be unreasonable, then a structural index may be obtained as in the case of columns and integral panels. The results are also similar except that titanium alloys appear to be superior to Stainless W at temperatures up to 400°C for most values of structural index. The relative weights of the plates are of the same order as those obtained by taking buckling as the limiting case.

### 4.3. PLATES SUBJECT TO SHEAR

Heimerl and Barrett[18] also consider the question of plates in which resistance to shear buckling is the criterion. Since, in plate analysis, the value of the reduced modulus is determined mainly by the secant modulus, this is used to facilitate computation and it is shown in the report that the same curves may be used for shear and compressive buckling; the results are essentially the same as those discussed for plates in compression.

### 4.4. CREEP

The major effect of extended exposure while the element is subjected to stress is that of creep deformation. The effect on tensile elements can cause large permanent deformations or even failure to occur after various times under load, depending upon the stress level and temperature. The behaviour of compression elements, however, presents a different picture since after a certain time under load an unstable condition will be reached.

4.4.1. *Creep Buckling*

As a simple example of a compression element, consider the case of a strut operating under a constant load, below the critical value and at a temperature such that creep is possible. The initial imperfections present in all practical structures will produce a slight lateral deflection which without creep effects would remain stable. The curvature, however, means that a non-uniform stress distribution exists and the effect of creep will be to cause the curvature to increase. The increase in curvature causes an increase in the difference of the stress levels existing in the strut, thus causing an increase in creep effects and eventually collapse will occur. It can be seen that under such conditions the column will collapse in time even though the applied load may be considerably less than the normal critical value. For columns designed for use at elevated temperature the term critical load is meaningless and applies strictly to short duration loading only, a load-critical time relationship taking its place.

The non-linear equations and irreversible plastic deformations which occur complicate the analysis of creep buckling; however, the problem has been treated at some length by Hoff[22, 23]. A numerical example considered by Hoff is a 24S-T3 aluminium alloy strut with a slenderness ratio of 111 which is stressed to 4,500 lb./in.$^2$ (the Euler stress is 5,900 lb./in.$^2$) at a temperature of 600°F. For this set of conditions failure occurs after 43 minutes under load. A further example[23] shows that the average applied stress has a significant effect upon the lifetime of the column and a small increase in area will have a beneficial effect. A 12½ per cent increase in area will, for example, increase the lifetime to about 110 minutes, a 256 per cent increase.

The effect of creep in aluminium alloy plate elements subject to compression has been investigated experimentally by Mathauser and Deveikis[21] and the results show that it is possible to estimate the stresses which are required to produce failure in a given time. The equation is of the same form as that for short duration loading, provided that time dependent, or isochronous, stress strain curves are used to derive the material constants.

A further effect of creep is to cause a redistribution of stress leading to a more even distribution, this applies to redundant structures and the various forms of stress concentration. The analysis of these effects is again complicated by the laws describing creep behaviour and has been considered by Hoff[23].

## 5. Thermal Stress (the Effects of Temperature Gradients)

The discussion in the previous section centred on the condition of a uniformly high structure temperature. These conditions apply to structural components in many cases, but when complete structures are considered, or when the flight conditions are changed, temperature gradients arise due to heat transfer within the structure. The general effect, in an accelerating aircraft, is to generate compressive stresses in the outer portions and tensile stresses internally, this being due to the rapid response of the outer shell which is in immediate contact with the heating source. A secondary effect is due to the reduction in stiffness with temperature which causes a change in the effective structure and, hence, a redistribution of loading. In this section the various effects of temperature gradient are illustrated by a series of examples and to simplify the treatment it has been assumed that the structures examined contain no fuel and are not being cooled internally, the temperature gradient being due solely to the heat conduction from the boundary layer.

### 5.1. PRIMARY (OR ENGINEERS' THEORY) THERMAL STRESS

In the analysis of structures it is often satisfactory to obtain the basic stress distribution by use of the Engineers' Theory of Bending and the Bredt-Batho Theory of Torsion in which certain simplifying assumptions are made. It is well known that the results obtained are satisfactory except in the regions of major discontinuities. A basic thermal stress distribution can also be readily obtained by making similar assumptions, and is subject to similar limitations. Thus, if it is assumed that plane sections remain plane and that $E$ and $\alpha$ are not functions of temperature, a simple expression for thermal stress may be obtained. For example, if the structure consists of one material only and is completely symmetrical the result is: —

$$\sigma = E\alpha\left(-T + \frac{\int_A T\,dA}{A}\right) \quad . \quad . \quad (10)$$

As an example, consider the case of a multi-web box structure which could be representative of the wing structure of a supersonic fighter. Fig. 7 shows the stresses at the centre of the web and in the skin midway between the webs. The three forms of construction chosen for analysis were all steel, all dural and steel skins with dural webs; in each case the proportions

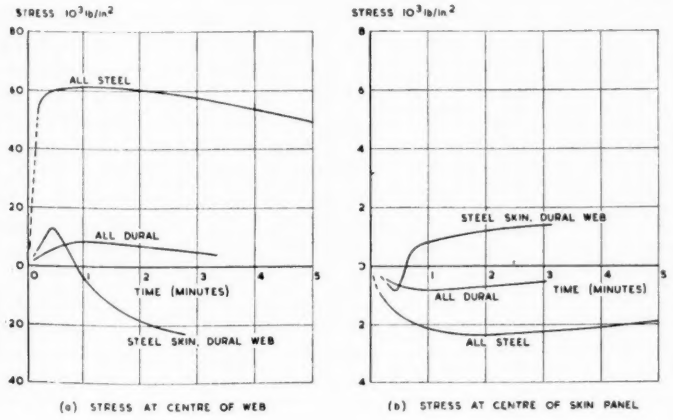



FIGURE 7. Thermal stresses in multi-web wings.

were adjusted to give an approximately optimum structure for a given loading, depth and torsional stiffness. The stresses arise from considering an instantaneous change in speed to $M=2{\cdot}5$ at 30,000 ft., after a long period of flight at $M=1{\cdot}25$ at the same altitude.

In the all-steel structures the thermal resistance of the thin webs, together with the small heat capacity of the skin, result in a large temperature difference between the web centre and the skin which, combined with a relatively high value of $E\alpha$, produces the large value of 60,000 lb.in.$^2$ tensile thermal stress at the centre of the web. The all-dural structure is subject to much less severe stress since the webs have much better heat conducting properties and $E\alpha$ is about 70 per cent that of steel. If a dural web is used in conjunction with a steel skin the higher heat conductance in the webs allows these to reach the skin temperature rapidly and the much higher value of $\alpha$ soon reverses the direction of the stress in the web; the thermal equilibrium state is one of steady temperature stress. The skin stresses shown in Fig. 7(*b*) are small by comparison due to the large skin area; these values must be regarded as approximate since the difference between skin and average temperatures amounts to about 4 per cent of the skin temperature and all temperatures were obtained by the electrical analogue computer.

The effect of the geometry of this type of structure on the thermal stress distribution has been considered by several writers, for example, Parkes[24] and Goldberg[25]. Their results may be summarised as follows:

(1) Increasing the ratio of web to skin thickness does not appreciably change the temperature distribution but increases the skin stress and decreases the web stress.

(2) Increasing the ratio of skin thickness to web pitch has a similar effect and increases the time to reach the peak stress since the thicker skins heat more slowly. Small variations do not significantly affect the stresses.

(3) Increasing the ratio of the web depth to web thickness results in a slight increase in web stress and a relatively large increase in the smaller skin stress; the time to reach the peak stress is also increased.

The material used has two effects, the first being due to the resistance to dimensional change produced by temperature and is determined by the product $E\alpha$ and the second due to the conductivity which controls the magnitude of the temperature gradients. The major requirements for the web are that it should respond rapidly to the heat flux from the skin and that it should offer minimum dimensional restraint to the skin. These conditions are ensured if the thermal diffusivity is high (*i.e.* high conductivity, low heat capacity) and if the modulus of elasticity is low and the coefficient of expansion is high. However, if the coefficient of expansion is too large compared with the skin, there may be high compressive stresses in the web, as previously illustrated.

Similar considerations apply to other types of stiffened skin panels. In actual structures there are naturally several other factors which modify these effects. For example, the effect of conductivity across joints, which is frequently low in riveted and bonded joints, will increase the difference in temperature between the skin and the web thus increasing the stresses. Griffiths and Miltonberger[27] have given the effects for a wide range of parameters. There is also the possibility of slip in the joint which would naturally reduce the stresses; an allowance in the analysis would be almost impossible in this case.

There may be appreciable effects due to heat transfer within the confines of the structure which may be either natural or deliberately introduced. For example, there may be fuel tanks or cooling which increase the temperature gradients, and heat transfer by natural convection or radiation may also occur which tends to relieve the gradients. These latter factors complicate the analysis and their effects have been examined recently by both Bisplinghoff[6] and Hoff[5]. Natural convection will in general be negligible but radiation may have a considerable influence if the skin temperatures are greater than about 300°C and low conductivity materials such as stainless steel or titanium are used.

### 5.2. CONSTRAINT STRESSES AND REDISTRIBUTION OF LOAD DUE TO CHANGE IN $E$

The effect of an increase in temperature is to decrease the value of $E$, hence, in the presence of temperature gradients, the effective structure is slightly different to that existing in the case of constant temperature. The magnitude is in general small and tends to increase the load carried by cooler parts of the structure, that is, in the interior during acceleration and in the skin during retardation. In addition, structural discontinuities may also cause substantial errors in the elementary stresses obtained above, in the same manner as those produced in the application of the Engineers' Bending Theory to these regions.

As a simple demonstration of these effects consider the case of a six boom tube having closely spaced rigid diaphragms in which the booms carry all the direct stresses. The loading consists of a force of 12,000 lb. applied at the centroid of the free end parallel to the webs, and it is assumed that the spar boom temperature is 120°C, while that of the stiffener is 170°C. The material is assumed to be aluminium alloy 75S-T6, and the variation of $E$ with temperature is considered; the flange and stiffener areas are 0·05 sq. ft., the skin thickness is 0·045 ft., the depth is 6 in. and the chord 20 in.

The stresses which exist in the top cover of the box are shown in Fig. 8, the dotted lines showing for comparison those calculated by the elementary theories. In these diagrams the contributions of the applied load and thermal stresses are shown together with the total stresses.

The boom stresses due to the applied load depart from the Engineers' Theory values due to two effects. First at the root, shear lag increases the flange stress and this is combined with the second effect which is the reduction of $E$ with temperature, thus making the central

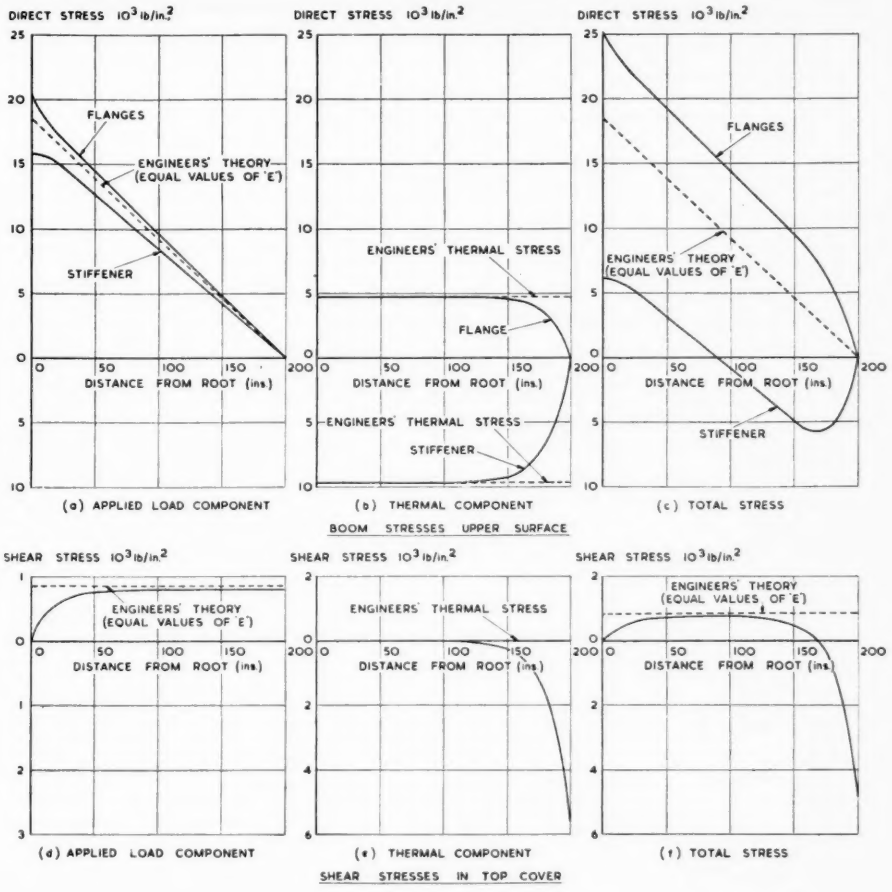



FIGURE 8. Stresses in the top cover of a six-boom tube.

stiffener less effective and this is of course evident over the whole length of the beam. The thermal stress shows no great departure from the elementary theory except within about two chords of the tip where the stresses decay to provide the stress-free tip condition. The change in $E$ with temperature is allowed for in the elementary thermal stress. The marked departure from the Engineers' Theory stress, due to the thermal component, is clearly shown in the combined stress diagram; the magnitude of the thermal stress in the stiffener causes a stress reversal in the outer half of the beam.

The effects on the shear stress in the plate are similar, the actual values being less than the Engineers' Theory since the stiffener, having the higher temperature, becomes less effective.

The small effect of the changes due to the variation in $E$ will be true in many practical cases. The majority of wing structures will depart considerably from this simple example and spanwise variation of temperatures, non-rigid diaphragms and non-symmetrical temperature and material distributions may arise in practice. Simplified cases of this nature have been considered by Heldenfels[28]. The possibility of distortions due to thermal stress coupling with aerodynamic forces to produce thermo-aeroelastic effects also needs consideration in the design stage.

### 5.3. THE EFFECTS OF THERMAL STRESSES ON TORSIONAL STIFFNESS

A further important effect of thermal stress distribution is to reduce the effective torsional stiffness of a wing structure, this being in addition to that due to the loss in the rigidity modulus. As as example consider a three per cent thick double wedge steel wing section having a 36 in. chord. Initially the conditions are steady temperature corresponding to a Mach number of 0·75 and 50,000 ft. and the section is then accelerated instantaneously to steady flight at $M=3$ at the same altitude. The temperaure distribution for this flight, allowing for the variation in heat transfer coefficient with distance from the leading edge (transition is assumed to take place in the range 5 in. to 10 in. aft of the leading edge) and chordwise conduction of heat, was obtained again by the electrical analogue. The thermal stress distribution shown for two values of time, in Fig. 9(*c*) is based on the elementary theory and shows very high compressive values at the edges of the wing, this being due to the rapid response of the thin sections. This high compressive stress has two possible effects which may lead to aeroelastic troubles. Firstly, it may cause the edges to buckle and secondly, any rotation of the section will produce a force component in the direction of the applied torque and thus cause further rotation

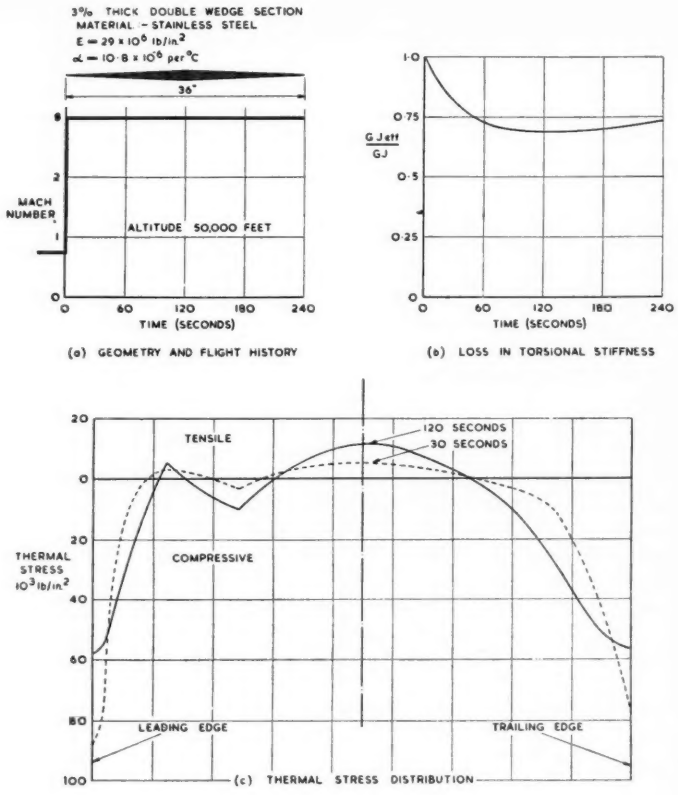



FIGURE 9. The effect of thermal stress on torsional stiffness.

reducing the effective torsional stiffness; this is shown in Fig. 9(*b*). More detailed work[26] has shown that the order of the reduction is not appreciably affected by the use of practical values for acceleration, or by consideration of the case of finite aspect ratio, unless these are very small. The effect is naturally affected considerably by chordwise mass distribution and is much less severe for hollow sections.

There are many other cases of thermal stress which could be considered, but it is hoped that some of the major effects have been demonstrated. A further case which is worthy of mention is that of skin panels supported by relatively heavy frames and in this case, as will be evident from the foregoing examples, there is the possibility of high compressive stress leading to buckling; this is most undesirable as there is then a grave risk of panel flutter (*see* Y. C. Fung[29]).

The calculation of thermal stress is somewhat involved in many cases but methods of analysis are readily available. For example, the normal skin-boom approach may be modified to include a thermal displacement term as in the tube example, and the extension of the conventional Energy Methods has been considered by Argyris[30], thus enabling the structural engineer to use well established methods.

## 6. The Alleviation of Thermal Stress

Having discussed the severity of the problem it is appropriate to consider now some of the means by which the problem can be reduced.

There is a considerable gain to be obtained from the aerodynamic aspects of the problem. A laminar boundary layer, as is well known, reduces drag and thus the heat transfer coefficient also, since the two are related. As the values are approximately one-eighth of those in a corresponding turbulent boundary layer and the recovery factor is also slightly reduced, the importance of the maximum possible extent of a laminar boundary layer is obvious. The effects of altitude are also of great importance since, at a Reynolds number of $10^7$, the heat transfer coefficient at 100,000 ft. is about one-eleventh of the value at 50,000 ft.

The leading edge of a supersonic wing presents a serious heating problem especially if a sharp edge is used as is often the case in missile design. The sharp edge has little mass and is subjected to an intense heat flux due to the high viscous shear in the very thin boundary layer, with the result that the edge temperatures rapidly approach the equilibrium value. The heat transfer coefficient is, in theory, inversely proportional to the leading edge radius, hence, increasing the leading edge radius reduces the heat flux and increases the mass, thus reducing the temperature response. The leading edge drag is increased but can be reduced by sweepback which again reduces the heat transfer coefficient.

Under extreme conditions a considerable relief may be obtainable from radiation and it appears that surface emissivities of the order of 0·8 should be possible, but the absorptivity may also be high and solar radiation under certain conditions will become important.

A considerable relief in temperature gradients may be obtained from the use of external insulation, the effect of which is to provide a high resistance to heat flow from the outer surface. This results in a rapid temperature response at the surface and a consequent reduction in heat flux. This reduction in heat flux increases the time required by the structural skin to reach a given temperature, thus insulation is particularly effective in short duration supersonic flight, *e.g.* in many missiles. For a given thickness the parameter thermal diffusivity, $k/\rho c$, defines the effectiveness of the insulator, the lower the diffusivity the greater the resistance to heat flow. However, the density alone is also important since, for a given weight, a thicker layer of an insulator having a higher diffusivity but lower density may be more effective.

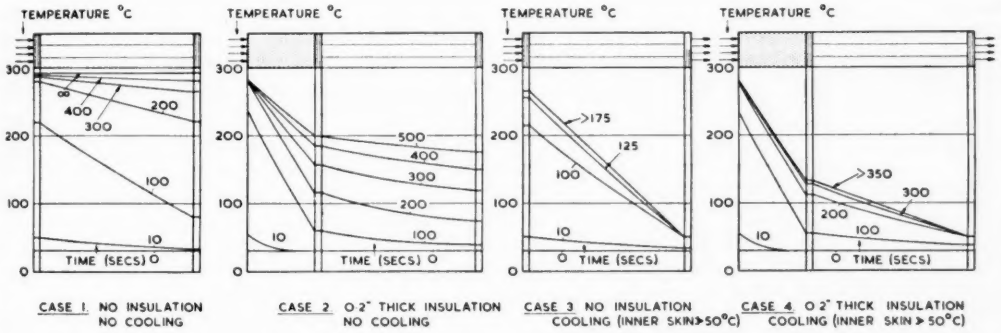
As an example, Fig. 10 shows the effects of combinations of insulation and cooling on the temperature gradients and thermal stresses in a stainless steel honeycomb sandwich panel. It is assumed that this is accelerated from $M=1{\cdot}5$ to $M=3{\cdot}0$ at 40,000 feet in two minutes, the speed then remains constant. The insulation consists of a 0·2 in. thick layer of plastic foam having the thermal properties of "Sebalkyd." The insulator is subject to severe conditions, namely, a high surface temperature, which in this example would actually rule out the use of "Sebalkyd", and high temperature gradients. It must also be bonded to the structure and resist aerodynamic forces and erosion.

Comparing cases 1 and 2 in Fig. 10 it can be seen that the insulation reduces the thermal stress to less than a third of that in the bare structure and increases the time required to reach the peak value. However, the structure temperature continues to rise and the inner skin approaches 200°C after ten minutes.

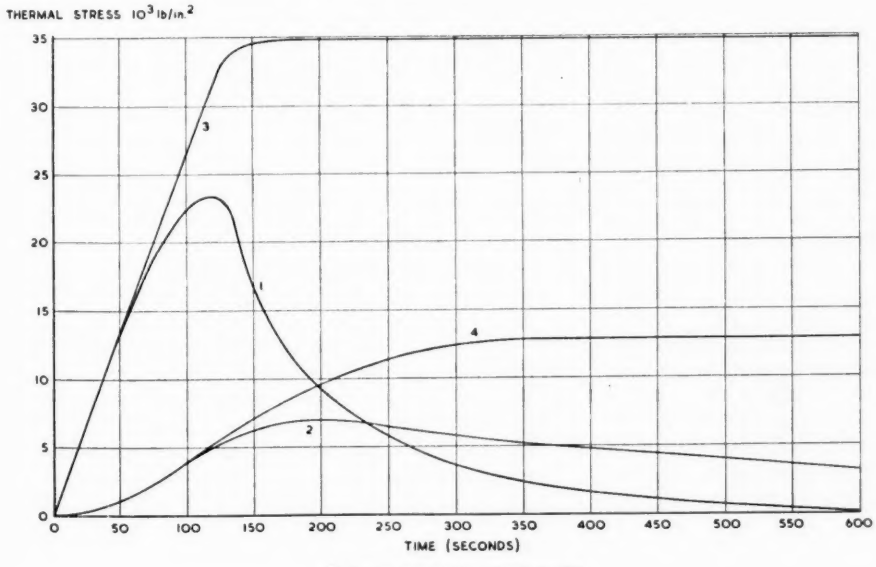
The structure temperature is reduced by using cooling but this alone is insufficient since a stable condition of high thermal stress is soon reached, as shown by case 3. The use of insulation and cooling is necessary to achieve a satisfactory solution for long flight, *see* case 4. The result is that after a time a steady thermal stress, almost twice that of case 2, is reached, but the inner skin never exceeds 50°C and the cooling required is about a fifth of a kilowatt per square foot which is just over a third of that required for case 3.

In many cases the presence of large quantities of fuel in contact with the skin provides an effect similar to that of cooling, but as the flight continues the temperatures begin to rise more rapidly as fuel is consumed. There is also the possibility of using the fuel as a heat sink in a cooling system; this however, needs very careful consideration in the design stage as the fuel systems engineer also has a share of temperature problems.

It may be possible to use a system in which a fluid coolant is injected into the boundary layer and come to grips with the problem at its source. Many advantages have been shown by theoretical work and it may be possible to obtain some control for short durations without carrying excessive amounts of coolant. There would, of course, be many practical difficulties in obtaining a satisfactory system.


TEMPERATURE °C
TIME (SECS)
CASE 1 NO INSULATION NO COOLING
CASE 2 0·2" THICK INSULATION NO COOLING
CASE 3 NO INSULATION COOLING (INNER SKIN>50°C)
CASE 4 0·2" THICK INSULATION COOLING (INNER SKIN > 50°C)

(a) TEMPERATURE GRADIENTS




(b) THERMAL STRESSES

FIGURE 10. The effects of insulation and cooling on thermal stresses in a sandwich skin.

Finally the structural engineer can minimise the effects by designs which reduce the temperature gradients within the structure and the restraint of the skin to thermal expansion. The joints to the skin form discontinuities in the heat flow path which increase the temperature differentials; reduction of these effects may be more easily achieved with steel structures in which welding or brazing may be a necessity.

The reduction in the restraint of the skin can be obtained by several methods and the system used will probably depend upon the overall design considerations. For example, in a multi-web box requiring closely spaced webs it may be necessary to reduce the thermal stress in the skins; this can be achieved by replacing some of the webs with flanges and post stiffeners. The restraint may also be reduced by using a corrugated web attached to the skin by a number of discrete cleats. Other means of alleviating the stresses due to internal structure which may be used are reduction of the internal redundancy by truss type elements in place of shear webs, or by the careful use of lightening holes in the web to reduce its extensional stiffness. A further possibility is the use of different material for the internal structure which has higher coefficients of thermal conductivity and expansion and a lower elastic modulus than the skin material. However, this must be carefully considered since, in some cases, the resulting temperature stresses may be large as in Section 5.1.

## 7. Design Requirements

The increase in structure weight resulting from the effects of kinetic heating and the large growth factors involved require that the supersonic aircraft, to attain the maximum efficiency, should be carefully tailored to its particular role. It is imperative that any design requirements must be realistic and comprehensive and to obtain the required high level of reliability, many of the official specifications will require revision. Furthermore, a large amount of test work will need to be undertaken to provide the basic design information.

Considering the latter point first, the effects of elevated temperature on materials and structural elements has been discussed, using averaged test results in most cases. The consistency of these results and the establishment of some form of specification minimum standards for a range of suitable structural materials, appears to be a necessity. For many designs it may also be necessary to establish some standard for creep performance on the lines already used in engine materials and, since this requires lengthy test programmes, an early start is necessary. It will also be necessary to establish the effects of alternating and cyclic loadings and also, temperature cycling upon the creep of materials. The experimental results obtained so far indicate that a form of cumulative damage law may be applicable; this has been discussed by Gerard[51]. There is also the interaction between creep and fatigue and it appears that a simple superposition of effects may not be entirely satisfactory.

The minimum design requirements must be clearly defined to obtain an optimum structure, and this will require some extension of the current flight envelope. In order to account for the combination of stresses due to manoeuvre and temperature gradients, the flight envelope requires generalisation to give the design inertia factors at given Mach numbers and altitudes under given atmospheric conditions. The maximum required flight times under high speed conditions also require specification and, in some cases, the most severe flight programmes anticipated could be included as an aid to efficiency. This data will also be required by other engineers, for example, those concerned with fuel and cooling systems. With this data it is possible to establish the design conditions of combined manoeuvre and thermal stress, together with the relevant material properties.

In some designs it will be necessary to consider the usable life of the structure as governed by creep. No general creep damage requirement can be specified since its actual form will depend upon each particular design. The limiting factors may be distortions involving a deterioration of performance or control characteristics, or alternatively local rupture or creep buckling may need consideration. Civil and large military aircraft are most seriously affected since a longer lifetime is required at relatively high stress levels due to the reduced manoeuvre factors. In assessing life time a detailed knowledge of the times at the various temperatures and loads is necessary, together with data on the cumulative effects of creep; Gerard's proposal appears to be satisfactory as a first approximation. Since considerations of creep can appreciably affect the allowable stress it will be necessary to formulate a realistic creep requirement for each design.

## 8. Conclusions

An attempt has been made to cover the fundamental aspects of the problem of kinetic heating and some methods of its alleviation. It will be apparent that, in view of the many parameters involved, any generalised statements must be regarded with some suspicion. However, in spite of the severity of the problem, it has been shown that there is enough data to enable the design of structures required to operate at moderately high Mach numbers to be approached with some confidence.

The need for much development work before the situation can be regarded as completely satisfactory is also evident and several topics for such work leading towards a more complete understanding of material behaviour at elevated temperatures have been mentioned.

The time available has prevented a full discussion of the problems involved in the manufacture and testing of the structure. On the production side much development work in the handling of the new materials, such as titanium alloys and stainless steel, will be required. The testing of structures under representative thermal and manoeuvre conditions will demand the use of extensive and costly test equipment to provide controlled and co-ordinated heating and loading cycles. The power needed will be at the rate of several kilowatts per square

foot and in many cases it will also be necessary to supply internal cooling to various parts of the structure. These aspects of test work are being studied by the Royal Aircraft Establishment and have been the subject of a paper by Horton[32].

The challenge of kinetic heating, unlike that of the so-called "Sound Barrier", is continuous and increases in severity as speeds are increased. The deterioration of suitable structural materials with increasing temperature will confine flight at high Mach numbers to the stratosphere where the heat transfer coefficients and hence, the heating rates, are relatively small. The structural problems are already being met and overcome. An example of this is the Bell X-2 Research Aircraft in which extensive use of stainless steel and K Monel is made in the primary structure. The most severe example under investigation is the re-entry phase of the Intercontinental Ballistic Missile and McLellan[33] has shown that the kinetic heating would be sufficient to melt a steel nose approximately one inch thick. Such research work will ultimately lead to a better understanding of kinetic heating and the behaviour of materials of elevated temperature and will enable the speed range of more conventional aircraft to be extended.

## Acknowledgment

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# Flight Development of the Avro CF-100 Mark 5 Aircraft

by

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SUMMARY: Improvements made to the aerodynamic characteristics of the wing of the Avro CF-100 Mark 5 aircraft are discussed. Modifications to the wing included increase in aspect ratio, addition of vortex generators, and deflection of trailing edge plain flaps. The effect of flap deflection and addition of vortex generators on the aerodynamic characteristics of the wing are shown to be closely associated with interaction of the upper surface shock wave with the boundary layer. Performance gains were demonstrated at high subsonic speeds at high altitudes. Vortex generators improved the buffet boundary, whereas flap deflection both increased aircraft ceiling and improved buffet boundary.

## 1. Introduction

Mention has been made in the Technical Press[1] of modifications to the Avro CF-100 twin-jet fighter which gave substantial improvements to aircraft performance. Partly because the subject has some intrinsic interest, and partly because of its Canadian association, some of the technical aspects of the problem are described here.

NOTATION

| | |
|---|---|
| $M$ | Mach number |
| $n$ | load factor |
| $p$ | static pressure |
| $W$ | aircraft weight |
| $\frac{1}{2}\rho V^2$ | dynamic pressure |
| $\sigma$ | pressure rise coefficient |
| $\omega$ | ratio of ambient to S.L. static pressure |
| $C_L$ | lift coefficient |
| $\gamma$ | ratio of specific heats |

*Suffices*

| | |
|---|---|
| (0) | refers to free stream conditions |
| (1) | refers to conditions immediately upstream of shock |
| (2) | refers to conditions immediately downstream of shock |

## 2. The CF-100 Mark 5

Reference One describes some structural modifications to the CF-100 which were designed to improve the performance of the aircraft and, in particular, raise the service ceiling. A concerted drive in the field of weight control resulted in reduction of the operational weight by nearly 4,000 lb. in spite of an extension to the outer wing panels. This was accomplished by removing all non-essential items and by a reduction in the fuel capacity. Outer wing panels were added at the wing tip which increased the span by approximately seven feet. The resulting increase in both wing area and aspect ratio were designed to give a reduction in drag (particularly induced drag) at high altitude and a corresponding increase in service ceiling.

The aircraft which incorporated these modifications is known as the CF-100 Mark 5, and substantial performance gains were demonstrated when compared with its predecessor, the Mark 4.

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### 2.1. MORE SUBTLE MEANS

Even before the CF-100 Mark 5 reached the flight test stage members of the Flight Research Section, N.A.E., and the Flight Research and Development Department at Avro had been considering other ways in which the ceiling performance might be improved. The N.A.E. embarked upon a research programme in which vortex generators were fitted to the wings of a CF-100 Mark 4 in an attempt to increase the lift available at high subsonic speeds. Later, tests were made at Avro with wing trailing edge and nose flaps deflected in an attempt to increase lift and reduce drag. Both programmes resulted in some degree of success.

## 3. Vortex Generators

Vortex generators are small stub-like wings placed in rows spanwise, usually on the upper surface of a wing. They are often made of aluminium strip bent into an angle to facilitate installation on the wing surface. Vortex generators stir up the boundary layer air and, in this way, alter the character of the interaction which takes place between a normal shock and the boundary layer on the wing surface at transonic speeds. The physical mechanism of the vortex generator has been described by Nonweiler[2], where he shows how the "bent tin" type of generator has been developed from the more sophisticated ramp type generator (*see* Fig. 1). The "dead air" in the boundary layer is swept up the ramp and vortices shed from the edges of the ramp supply clean air to the wing surface between each generator. It was found later that the simple "bent tin" type of generator, by shedding small vortices at the tip, could scavenge and rejuvenate the boundary layer in a similar way. The "bent tin" variety are placed at approximately 15 degrees to the oncoming stream and may be co-rotating or contra-rotating, this referring to the direction of the shed vorticity (*see* Fig. 2).

## 4. Interaction of Shock Wave and Boundary Layer

It is well known that shock waves form on the upper surface of a wing at high subsonic speeds. The shock is nearly normal to the wing surface and moves towards the trailing edge as the Mach number increases. Study

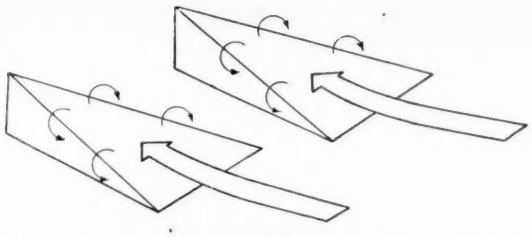

FIGURE 1. Boundary layer air being carried up ramp type generators. Shed vortices scavenge boundary layer and promote mixing.

of the interaction of the shock with the boundary layer by Holder, Pearcy, and Gadd[3], and the nature of this shock movement by Küchemann and Hartley[4] has revealed two interesting phenomena.

First, it was noted that the pressure ratio across the shock was nearly constant and independent of the free stream Mach number, $M_0$. Likewise, the local Mach number upstream of the shock was nearly constant. These two facts are, of course, related by the nature of shock waves and the local Mach number downstream of the shock is then also independent of $M_0$. The explanation of this was found in the nature of the boundary layer. The flow external to the boundary experiences a sudden increase in pressure through the shock wave, but the boundary layer flow is mainly subsonic and contains no discontinuity through which the pressure may rise. To establish a stable flow the pressure in the boundary layer must rise to match that of the external flow and this can be accomplished only through mixing of the two flows. Thus Küchemann[5] introduces the physical hypothesis:—

"that the pressure rise through the shock in the external predominantly inviscid stream at the edge of the viscous region is limited by what the viscous region itself can muster by means of turbulent mixing, such that the pressure rise coefficient

$$\sigma = \frac{p_2 - p_1}{\frac{1}{2}\rho_1 V_1^2}$$

does not exceed a certain value which may, among other things, depend upon the state of the boundary layer (roughly, $\sigma$ is about one half for turbulent layers and about one third for laminar layers upstream of the shock). For example, if the shock lies in a plane normal to the main stream,

$$\sigma = \frac{p_2 - p_1}{\frac{1}{2}\rho_1 V_1^2} = \frac{4}{\gamma+1}\left(1 - \frac{1}{M_1^2}\right),$$

which gives for the upstream Mach number

$$M_1^2 = \frac{1}{1 - \left(\frac{\gamma+1}{4}\right)\sigma}$$

which for $\gamma = 1{\cdot}4$ gives $\quad M_1^2 = \dfrac{1}{1 - 0{\cdot}6\,\sigma}$

We then have $M_1 = 1{\cdot}2$ for $\sigma = 1/2$, independent of the main stream Mach number, $M_0$. This 'ceiling' of the upstream Mach number has often been observed."

It is, in fact, this condition of equilibrium that positions the shock wave on the wing surface. Eventually the boundary layer is unable to adjust itself to the pressure rise through the shock in the external inviscid stream and it separates from the surface. It is this interaction between shock wave and boundary layer which causes the layer to separate from the surface and as a rule causes the formation of a long separation bubble which extends beyond the trailing edge, thereby lowering the pressure at the trailing edge. Thus the circulation around the aerofoil, and with it the lift force, is not firmly fixed as it is in the absence of the bubble, and the result may be a low-frequency oscillation of the lift which is commonly called buffeting. This is the nature of the so-called "shock stall." It does not necessarily imply a maximum attainable lift, although in many cases it corresponds to the maximum usable lift. From such considerations come the terms "buffet lift coefficient" often denoted by $C_{L_B}$ and "buffet boundary" (Figs. 3 and 4).

The presence of vortex generators on the wing surface, by increasing the ability of the inviscid and

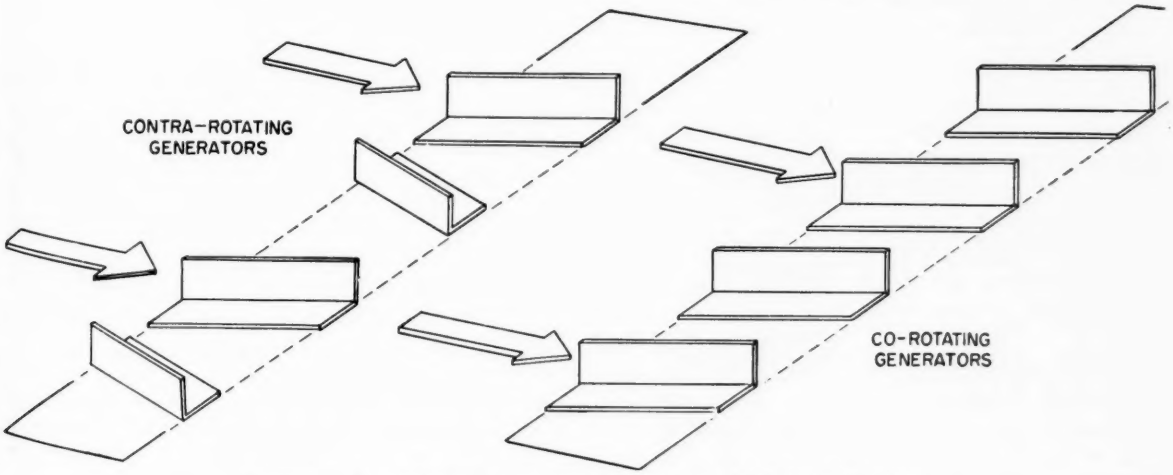


FIGURE 2. Co-rotating and contra-rotating generators of the "bent tin" variety.

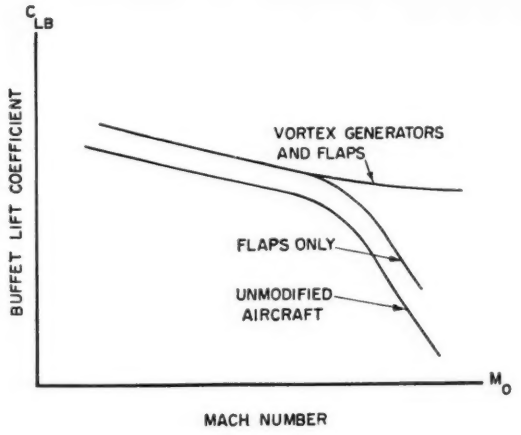


FIGURE 3. Buffet boundary $C_{L_B}$ plotted against $M_0$.

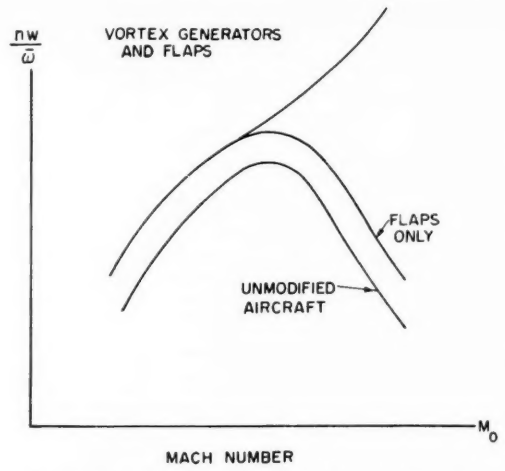


FIGURE 4. Buffet boundary $nW/\bar{\omega}$ plotted against $M_0$.

viscid layers to mix, permits a greater pressure rise through the shock and enables the wing to provide greater buffet-free lift. For the generators to be effective therefore, they must be ahead of the shock wave. A problem with respect to location of the generators then arises; if they are too far forward on the wing the power to mix may fade towards the trailing edge and the generators then tend to be ineffective at high Mach numbers; if they are too far from the leading edge the shock wave may be ahead of them and the generators tend to become ineffective at lower Mach numbers. One difficulty in design of vortex generators is that a large number of variables are involved; length to height, size, spacing, chordwise position, spanwise extent, and so on. One usually, therefore, arrives at a configuration which provides some improvement, but one is never sure that the configuration is necessarily an optimum.

The second interesting phenomenon noted by Küchemann and Hartley[4] was that velocities and pressure distribution on the surface of the aerofoil behind the shock are very similar to those of the "shockless" flow, where flow without shock is scaled from incompressible flow by, say, the Prandtl-Glauert rule. To quote again from Küchemann[5]:—

"Evidently, the shock accommodates itself in such a way that, at the boundary of the viscous region, the pressure rise in the external stream is compatible with the pressure rise through mixing in the viscous region and that the downstream pressure smoothly joins into a pressure distribution which is governed primarily by the then shockless external stream."

The significance of this phenomenon in relation to the CF-100 will be discussed later.

## 5. Vortex Generators on the CF-100

The Flight Research Section of the N.A.E. undertook a test programme with a view to understanding the nature of the buffet limit on the CF-100 and then to improve it by addition of vortex generators to the wing surface[6]. The author wishes to compliment the N.A.E. on a well conducted programme in the field of flight testing, where a truly successful test is often difficult to achieve. The full story of the work would best be told by members of their staff, suffice it to say at this time that the buffet was improved considerably at speeds above $M_0 = 0{\cdot}7$. The effects of shock stall on the CF-100 become quite appreciable above $M_0 = 0{\cdot}7$, as evidenced by a sharp fall off in $C_{L_B}$. The design and location of vortex generators were progressively modified by the N.A.E. until the buffet lift coefficient at $M_0 = 0{\cdot}7$ was maintained up to the design diving speed (*see* Fig. 3). The addition of generators altered flight characteristics in the following way: without generators the maximum "$g$" at constant altitude would rise and then fall as the speed increased; the addition of vortex generators enabled the pilot to apply more "$g$," and no peak in the curve was evident up to the design speed.

A method of presentation of results which helped to facilitate analysis was suggested by the author. The aerodynamicist plots buffet boundary as a variation of $C_{L_B}$ with Mach number and in this way eliminates the variables of aircraft weight and height. The pilot, however, experiences buffet boundary as a variation of "$g$" with speed at a given weight and height. A plot of buffet boundary in terms of $nW/\bar{\omega}$ against Mach number provides the best of both worlds in that it both eliminates the variables of weight and height, and also presents a curve of similar character to that experienced by the pilot. In addition, the computation required is less than in the case of the $C_L - M_0$ curve and, consequently, flight test results may be presented with the minimum of mutilation (*see* Fig. 4).

One further interesting result became apparent. It was found that the removal of a single generator from certain locations in the row could almost eliminate the beneficial effects of the configuration. This may account for the fact that use of vortex generators has met with limited success in some applications.

## 6. A Flight Test Programme at Avro

A flight test programme was made by Avro Aircraft Ltd. which aimed to reduce the drag of the CF-100 at high altitude and, by so doing, to increase the aircraft ceiling.

It was argued that increase in drag would probably be associated with some form of flow breakaway; if the flow breakaway could be cleaned up, then the drag could be reduced. The upper wing surface and nacelle on one side of a CF-100 aircraft were therefore painted white and wool tufts were located over quite a large area (Fig. 6). Ciné cameras were used to photograph tuft behaviour at high subsonic speeds and tests were made in steady turns at high altitude.

When the onset of buffet was encountered tufts on the wing became unsteady: this unsteadiness occurred aft of the 30 or 40 per cent chord point and extended across the wing almost to the tip. Before this, flow over the wing was remarkably smooth. On occasions, by approaching the buffet limit quite slowly, it was possible to induce unsteadiness in a single spanwise row of tufts while those ahead and behind the line remained quite smooth. It seemed clear that the unsteady tufts located the position of the upper surface shock wave, and that the pressure rise through the shock had caused separation of the boundary layer. In the case of gentle buffet, re-attachment of the flow occurred, and in this case, only those tufts in line with the shock wave remained turbulent. It was concluded that the wing had reached the limit of its capability and that only by modification to the basic wing characteristics could the buffet boundary be improved.

Reduction of drag in the flight régime below the buffet limit could, it seemed, be achieved only by modification to fairings such as, for example, the canopy or nacelle. The extent of improvement to be achieved in this way, however, was uncertain. Two attempts were made to obtain more favourable wing flow, first, by deflection of the nose flap and, secondly, by deflection of the trailing edge flap.

### 6.1. DEFLECTION OF THE NOSE FLAP

A nose flap, hinged on the lower wing surface, is fitted to the CF-100 and used to gain access to the wing structure. This was deflected through small angles and suitably faired to retain a smooth wing contour. Tests were conducted at high altitudes, but with no perceptible improvement.

### 6.2. DEFLECTION OF TRAILING EDGE FLAP

The trailing edge plain flap was deflected through ten degrees and, in this case, some improvement in performance was immediately apparent. Not only did flap deflection provide a means whereby the buffet lift coefficient could be increased, but it also gave a configuration which, for the same lift coefficient, showed a substantial reduction in drag.

The former result was anticipated and lift gains corresponded roughly to low speed values. Deflection of landing flaps generally, however, incurs an increase

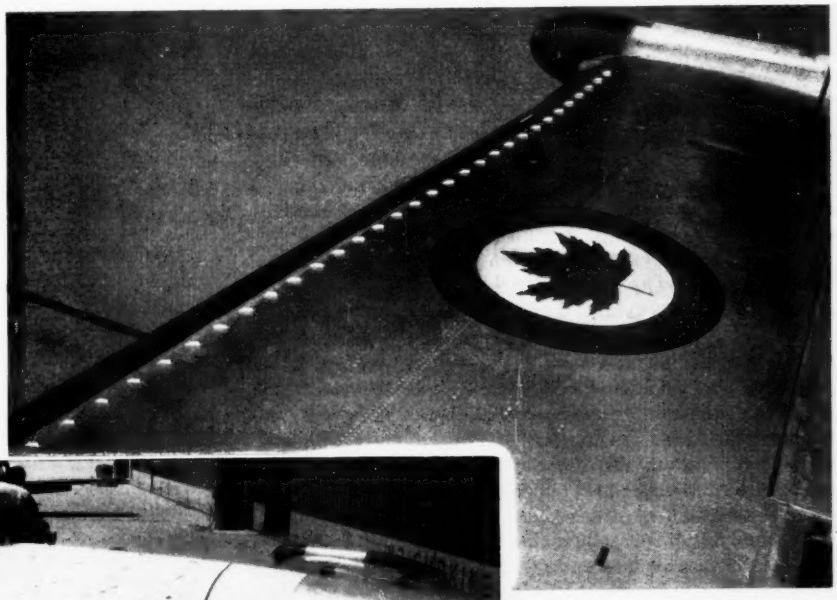

FIGURE 5 (*above*). Vortex generators on the wing surface of a CF-100 Mark 5 aircraft.

FIGURE 6 (*left*). The CF-100 Mark 5 aircraft being prepared for a test flight to investigate the nature of buffet at high speeds.

in drag, and the question naturally was: "Where did the reduction of drag come from?" This was a rather unusual situation; one is usually required rather to diagnose the increase in drag! The ceiling of the aircraft was increased by approximately 1,500 ft., and maximum level speeds at high altitude were also slightly improved.

### 6.3. THE DRAG REDUCTION

It was suggested that a change in spanwise loading, due to flap deflection could account for the reduction in drag or, again, that the reduction in angle of attack of the nacelle and fuselage would account for it. Whereas one cannot rule out such possibilities as contributing factors, it remained, however, for D. Küchemann of the R.A.E. to suggest, perhaps, the most likely explanation. He argued that, as in the case of a wing without flap, the velocity distribution over the rear portion of the aerofoil behind the shock wave would again correspond to that for a shockless inviscid type of flow. The "bump" in the velocity or pressure distribution due to flap would arrest the rearward movement of the upper surface shock wave and, comparing cases with and without flap at the same lift coefficient, we would find the shock located farther forward on the flapped wing. A reduction in wave drag would follow directly from the more forward position of the shock and, provided the increase in profile drag due to flap deflection was not very great, then an overall drag reduction would result. Again, it is possible that, within a certain lift range, a shock-induced separation may occur with "flap up," whereas this may be avoided with "flap deflected" when the shock can be farther forward. If this happened deflecting the flap would again tend to reduce the drag and postpone buffeting.

The suggested type of flow pattern is illustrated in Fig. 7, which shows the variation of pressure coefficient with and without flap deflection at the same lift coefficient. Corresponding values for shockless inviscid flow are also shown. Note that the upstream pressure coefficient (hence, also, the upstream local Mach number) and the pressure discontinuity due to the shock wave are the same for both the flapped and unflapped aerofoil. Note also how, in each case, the pressure coefficient behind the shock complies with inviscid flow conditions. These assumptions follow directly from previous discussion, and determine the position of the upper surface shock wave. It remains for experiment to check the validity of these ideas and perhaps exploit them more fully.

Flight tests provided an indication that the increase in profile drag due to a small flap deflection was itself small. Tufts on the upper surface of the trailing edge flap remained quite smooth with a deflection of only 5°, at 10° the tufts became turbulent, and at 15° they were quite unruly. It is reasonable to suppose that very little profile drag penalty would be incurred with 5° flap, an appreciable effect with 10° flap, and an over-riding effect at 15°.

The two effects described above conflict, and the overall drag falls and then rises again as the flap angle is increased. Flight test results were used to find an "optimum" position of the flap, that is, one just beyond the point of minimum drag, which then provided also an appreciable gain in lift available.

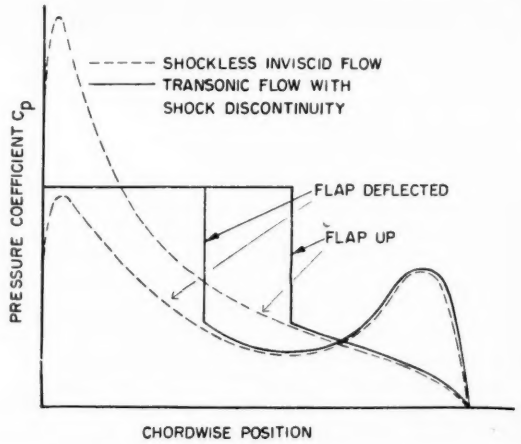


FIGURE 7. Idealised chordwise variation of pressure coefficient on upper surface with and without flap deflection at the same lift coefficient. Corresponding "shockless" inviscid flow shown dotted.

The foregoing considerations apply to flight at high subsonic speeds and at high altitudes.

### 6.4. COMBINATION OF VORTEX GENERATORS AND DEFLECTED FLAP

It was found that the two effects were additive and that an aircraft fitted with both vortex generators and deflected flap showed substantial improvement to the buffet boundary. Generally it could be said that the drag reduction due to flap was offset by drag of the generators.

## 7. Conclusion

The tests at the National Aeronautical Establishment and at Avro Aircraft Ltd. demonstrate the importance of flight research in the development of high speed aircraft. It is a costly means of testing but, given the necessary freedom, and with wise direction, can provide real dividends.

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# The Pressure Jet Helicopter*

by

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BEFORE enlarging on the main subject of this paper, it will be helpful to clarify some fundamental conceptions of various rotor drive configurations.

In the past the original shaft-driven helicopter presented two major problems to the designer. These were first: the necessity to choose between either some torque balancing device or a multi-rotor layout and, second: the necessity of a large reduction gear box, the weight of which seemed to become prohibitive with increasing all-up weight of the design. Big helicopters with either single rotor/tail rotor configuration or multi-rotor configuration, are flying successfully and prove that these afore-mentioned problems, although still in existence, were mastered in an admirable way. Nevertheless, in the old days the words "tip drive" had a magic sound to the harassed designer. No more reduction gears and heavy shafts, no tail rotor, much less rotor vibration due to the possible elimination of drag hinges; these were the big advantages claimed by the tip drive enthusiast. As I myself was one of them, I learned the hard way about all the major and minor problems which, at first sight, were overlooked in view of the big promises. Therefore I intend to spotlight those problems connected with the tip drive rotor in general and, in particular, those connected with the pressure jet with fuel burning at the tips and, by doing so, to show the limits inside which this type of rotor propulsion is superior to its fellow arrangements.

The tip-jet drive can be divided into two big branches, the pure tip-jet drive with the whole power plant attached to the tip of the rotor blades, and the pressure-jet drive with part of the power plant stationed in the fuselage, or in the centre of the rotor with rotor blades and tip propulsion units as parts of the propulsion system.

The pure tip-jet drives are:

the ram-jet,
the pulse-jet,
the tip-located jet engine.

Although not strictly inside the scope of this paper, a brief description is given for the sake of completeness. The ram- and pulse-jet drives are, from the power plant point of view, the most attractive propulsion system, because of their simplicity and low installed engine weight. Closer examination of the practical adaptability reveals a number of problems. With the subsonic ram-jet, the propulsion efficiency calls for a very high tip speed which, in turn, limits the forward speed due to the compressibility effect in the advancing blade or, by lowering the tip speed in forward flight and considering the low solidity of the ram-jet rotor, the tip stall in the retreating blade becomes a problem. Furthermore, due to the high drag of the ram-jet unit, the rate of descent in case of power failure is very high indeed. The high specific fuel consumption makes this drive attractive only for very short endurance, say under half an hour. For longer endurance, the main advantage of high disposal load is lost due to excessive fuel weight.

The advantage of the pulse-jet scheme against the ram-jet is the elimination of a rotor starting device, its lower specific fuel consumption and higher thrust output per unit frontal area. On the other hand, the highly stressed intake valve with a rather limited life of only a few hours, the long tail pipe and the low tip speed for optimum propulsive efficiency, are mechanical and aerodynamical disadvantages.

The tip-located turbine, being the most efficient pure-jet drive, may be attractive for the aircraft designer as a power plant for very big helicopters, although he has to confront the problem of blade bending of the stationary rotor and the high centrifugal loads on the blade structure and, in addition, only the future will show how the engine designer will master the problem of bearing loads and gyro couples of jet turbines working at a few hundred *g*'s at the tip of the rotor blades. Still, in view of the extreme attraction of this layout for the giant helicopter, a solution for these engine problems may be found.

The second branch of the tip-jet drive, *i.e.* the pressure-jet drive, is a more complicated scheme.

The elimination of rotor clutch, reduction gear, drive shaft and torque balance of the single rotor shaft-driven helicopter, is followed by the introduction of an engine-driven compressor, a ducted head with rotating seals and a ducted blade hinge arrangement, ducted blades and tip nozzles with or without burners. This considerable complication in comparison with the pure tip-jet drive is accompanied by much lower specific fuel consumption and better aerodynamic characteristics in forward flight and auto-rotation.

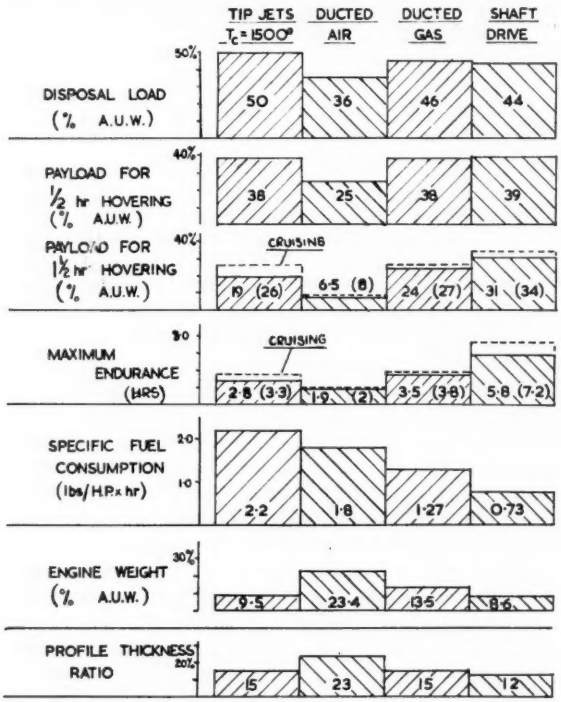
It is important to stress the fact that, as distinct from the shaft drive or pure tip-jet drive, in the pressure-jet-driven helicopter, the head, blades and tip nozzles are part of the power plant system and a modification of any of these members influences the design of the rest.

There are three basic configurations of the pressure jet drive:

(*a*) the ducted air scheme
(*b*) the ducted gas scheme
(*c*) the ducted air scheme with fuel burning at the tips.

Each of the three schemes has its advantages and only very thorough design study can determine which is the best for a particular type of helicopter. A number of papers (*see* References) deal with the merits of the various schemes and some excellent design studies

*A Section Lecture given on 9th October 1956.




Power plant : Turbo-compressor. A.U.W. : 30,000 lb.
Disc load : 6 lb./ft.$^2$ = Constant
Tip speed : Corresponding to optimum endurance

FIGURE 1. Comparison of parameters for various rotor drives.

and comparisons are available. Nevertheless, every general comparison of those three schemes, however accurate the assumption of the basic parameters, will be subject to high controversy about the selection of the common parameters. Here again, I would like to state that, according to my experience with a number of design studies, it is nearly impossible to obtain a fair comparison of merits and disadvantages by assuming any common parameter.

As already mentioned, the design of the pressure-jet propulsion system, which includes head and blades, influences to a high degree the aerodynamics of the rotor system, and this to a different degree in each of the three systems. Therefore a common assumption, either regarding engine size, or of aerodynamic parameter, must give a misleading comparison. The number of conflicting features in such a complicated power system are too numerous. There is the main conception of the aircraft, either as a pure hover plane or flying crane, or a fast-flying transport version with a small percentage of hovering time inside the total flying time, then the all-up weight, the expected disposal load, the payload, and the fuel economy regarding range and endurance. Closely attached to the performance requirements are the aerodynamic parameters like disc load, tip speed and rotor solidity. These parameters, dictating the dimensions of the blades, not only govern the necessary power as in the shaft-driven design but, with the ducted blade arrangement, they highly influence the efficiency and weight of the propulsion system.

From the foregoing it can be concluded that, for a set of basic performance requirements a separate design study for each scheme has to be made by changing the various parameters of the scheme under consideration until an optimum combination, acceptable from the performance, design, stressing and economic point of view, is obtained. Then these optimum solutions have to be compared for the final selection of the propulsion system. I would therefore like to restrict myself to a review, in general terms, of these three schemes, pointing out only the most striking differences.

Figure 1 shows a comparison of the characteristics of three pressure-jet systems with those of the shaft-driven helicopters. This comparison is based on a selection of common design and performance features. These were partly based on values taken from Ref. 3 and corrected according to experimental results regarding pressure and heat losses in rotor blade ducts and tip-jet units. The constants are: all-up weight, type of basic power plant, disc load, number of blades and cruising speed. The variables are: the compression ratio of the auxiliary compressor, the air speed through the ducted blades, the tip speed, the rotor solidity.

From the foregoing explanation, it is obvious that too many common parameters are assumed in the above study which, however excellent in its conception, must lead to controversy regarding the results. Nevertheless, it gives a good illustration of the outstanding features of the three pressure-jet drives. The choice of a turbine as a compressor driving power unit is rather obvious inasmuch as it eliminates the gears in a piston engine-driven compressor plant, which was always considered as a re-introduction of a feature which was proudly claimed to be overcome by the tip-jet drive conception. The lower weight/power ratio, the low frontal area, reduced engine vibration and the lower maintenance and inspection costs should out-weigh the higher fuel consumption.

In general, it is safe to state that only for helicopters from about 20,000 lb. all-up weight upwards, the pressure-jet drive gains advantage, with regard to complexity and weight of power plant and transmission, over the shaft-driven single rotor system for which the reduction gear and shaft weight increases considerably with the all-up weight. Nevertheless, selecting a special compressor plant where higher specific fuel consumption is accepted in return for simplicity and power weight ratio, shows the pressure-jet drive as an attractive solution even for very small helicopters like the Fairey Ultra Light and the Djinn helicopters.

Although the tail rotor as a torque balance can be eliminated, some sort of yaw control for hovering has to be re-introduced, a feature which is an unpleasant complication of the tip-driven large helicopter where the thrust for forward flight is supplied by a single main rotor. Here again, very small pressure-jet helicopters with low inertia in yaw (Fairey Ultra Light and Djinn) show a simple solution by using the exhaust of the Palouste turbine compressor plant for directional control.

The simplest pressure-jet drive is the ducted air scheme where no tip units and no tip-jet fuel system is

required. In view of the problems, described later, connected with these features, this drive is, at first sight, very attractive indeed; on the other hand, the efficiency with regard to the ratio of available rotor power to the installed engine power is very low and the value of this ratio is approximately one-third. Low compression ratios have to be chosen for propulsive efficiency. On the other hand, to accommodate the ducting space inside the blade section, together with keeping the c.g. of the blade at the aerodynamic centre, large chords and thickness ratios are unavoidable. Due to the big engine installation, heavy rotor hubs and blades, the disposal load is bound to be low and, as Fig. 1 shows, the high specific fuel consumption, short endurance and the inferior blade aerodynamics, makes this scheme unattractive from the performance and economy point of view.

Still there are applications where, with all these disadvantages, the simplicity remains the deciding factor. The Djinn is the example for this statement and shows that in a small aircraft by choosing a low disc load and designing a specially suited power plant, an acceptable compromise between performance and operating costs can be achieved.

The ducted gas scheme has much better efficiency than the pure air drive (*see* Fig. 1). For this particular design study the payload endurance characteristic for a prolonged hovering or cruising period compares best with the shaft drive characteristic. In this scheme the blade and head structure presents a number of difficulties due to the high gas temperature and subsequent loss in strength of material, which can only be overcome by very careful design. In principle this scheme still requires rather high rotor solidities and is specially suited for aerial cranes or medium speed helicopters.

The third variation, the ducted air scheme with fuel burning at the tips, on the assumptions of design study (Fig. 1) has the highest specific fuel consumption. On the other hand, it has the lowest empty weight and the hover endurance, being slightly lower than that of the ducted gas scheme, is appreciably better than that of the pure air drive. Although it is the most complicated version of the three systems, it gives the highest rotor power for a given engine power and is especially suitable for applications where high rotor power is required for comparatively short periods. This is the case for the convertiplane configuration, where the rotor is driven only during hovering and for starting and landing (*e.g.* Fairey Rotodyne). Another example of its application is the Fairey Ultra Light jet helicopter where a high rate of ascent can be achieved with the available surplus power.

As already mentioned, the rotor of a pressure-jet helicopter has to be considered as part of the power plant. In the case with tip burning, this fact is of the utmost importance. The rotor, being designed for a certain all-up weight and aerodynamic performance, dictates the selection or the design of the compressor plant to such a degree that, I maintain, to obtain best efficiency, the compressor has to be specially designed as part of the pressure-jet power plant. Compared with this requirement the selection of the engine driving the compressor is of secondary importance.

Having spent most of my work on pressure-jet helicopters with tip burning, I would like to enlarge on a number of problems connected with this type of drive where I can put experience before theoretical conclusions. The pressure-jet system with tip burning is the only drive where rotor power exceeding the engine power can be achieved. This feature is very important so far as multi-engine reliability is concerned.

In general, multi-engine reliability represents some difficulties in a pressure-jet helicopter inasmuch as in case of failure of one engine and without special precautions, the ratio of residual rotor power to normal rotor power is less than the ratio of residual engine power to normal engine power. Equal ratios can only be achieved either: (*a*) by adjustable tip jet nozzles, (*b*) by separate ducting from each engine to pairs of rotor blades, or (*c*) by a power plant arrangement, where the power plants are turbo-compressors driving free turbines, which in turn drive the common main compressor.

With the pressure-jet system with tip burning incorporating one of the installations described, an increased ratio of residual rotor horse power to residual engine power in case of engine failure can be achieved by increasing the fuel-air ratio in the tip burners above the normal rate.

The pressure-jet drive with tip combustion chambers permits the design of aerodynamically highly efficient blades and is therefore, whatever its application may be, especially suited for fast flying rotating wing aircraft. The optimum design parameters still depend on the operational application. Fig. 2 shows an illustration of the relations between compression ratio, specific fuel consumption and the ratio of rotor power to compressor shaft power. Such a diagram can be drawn for a combination of constants, based on a certain blade design which, in turn, should be based on aerodynamic requirements. In Fig. 2 this relation is shown for a tip speed of 750 ft./sec. with an assumed practical pressure drop in the blades based on blade design study. The other assumptions for the given relation are: the basic power plant is a turbo-compressor engine driving an auxiliary compressor with a compressor efficiency of 84 per cent, the tip-jet combustion efficiency is 90 per cent and the tip-jet drag coefficient is 0·10. It shows that to obtain high rotor power, rather high jet temperatures are required. Furthermore, for a constant fuel-air ratio the lowering of the compression ratio results in an increase of rotor horse power due to better propulsive efficiency. This relation, although in principle the same as on the air drive or ducted gas drive, is much less distinct due to the higher jet speed from the high temperature combustion chambers. From the fuel consumption point of view, it is obvious that weak mixtures are preferable and there is in each case an optimum compression ratio regarding fuel consumption. In this particular case (which is fairly representative due to its realistic assumptions) the optimum pressure ratio or the tip-jet drive is between four and five.

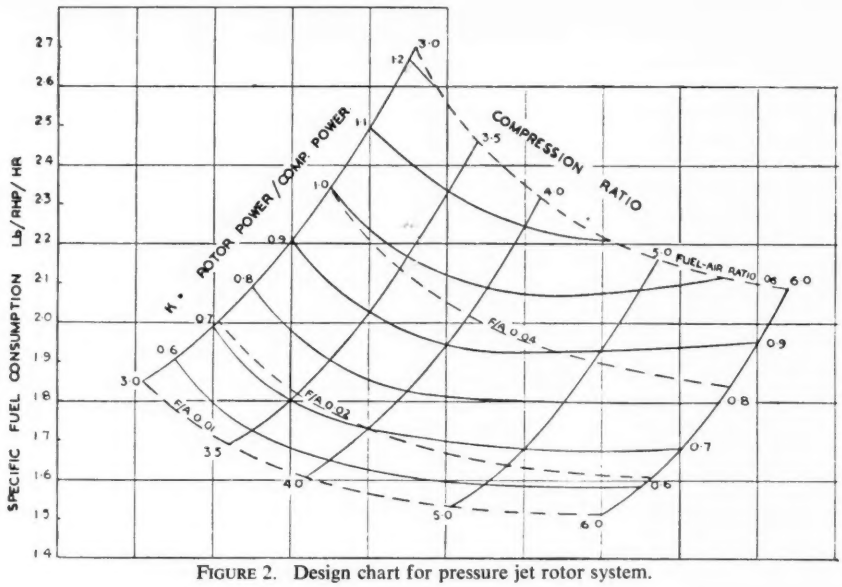



FIGURE 2. Design chart for pressure jet rotor system.

Another illustration of design parameter is shown in Fig. 3, this time for a piston engine-driven compressor with the following assumption: power transmission losses due to yaw control propellers, cooling fan and two-stage gear box = 18 per cent, compressor efficiency = 80 per cent, specific fuel consumption of the engine driving the compressor = 0·5 lb./h.p./hr., pressure drops in the ducts based on design studies, combustion efficiency = 90 per cent, tip-jet drag coefficient = 0·10. From these curves the influence of tip speed on power ratio and specific fuel consumption can be seen. The optimum pressure ratio is again around 4·0. The pure air drive is included. This is the point where the air-fuel ratio reaches infinity and the steep rise in specific fuel consumption and fall in power ratio for this combination of assumption is remarkable. Further, it can be noted that the optimum specific fuel consumption improves (although only a little) with tip speed, while the power ratio at optimum specific fuel consumption is very low indeed and rather independent of the tip speed. On the other hand, at rich mixtures with an assumption of a constant fuel consumption of, say, 2 lb./h.p./hr., there is a remarkable increase of power ratio from 0·67 at 650 ft./sec. to 1·07 at 800 ft./sec. corresponding to a change in air-fuel ratio from $R = 35$ to $R = 18$.

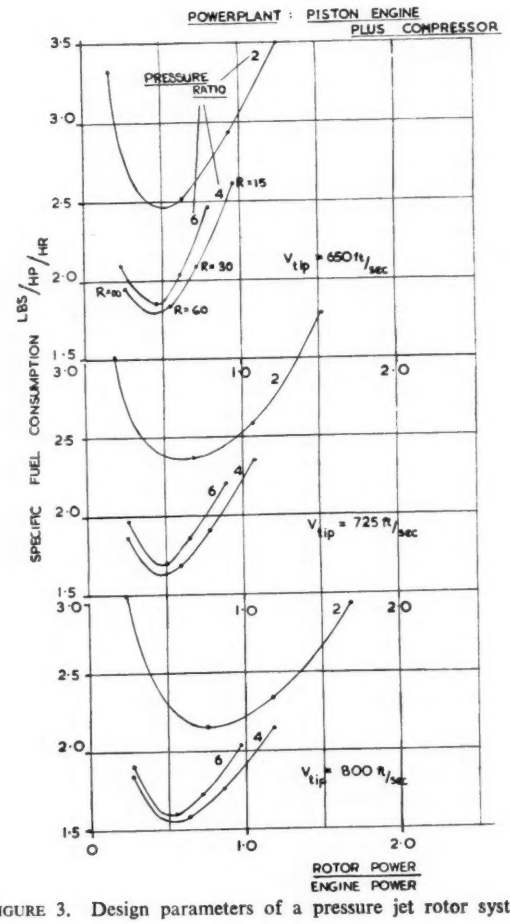



FIGURE 3. Design parameters of a pressure jet rotor system.

An interesting fuel balance is shown in Fig. 4 for a pressure-jet helicopter with two 870 h.p. piston engines driving propellers and compressors and with a rotor running at 800 ft./sec. tip speed at a net rotor power of 1,200 h.p. Fig. 4 shows a fuel balance at a compression ratio of 5 and 2. Although the theoretical consumption without losses is lower in the low pressure case, the power losses due to friction in the ducts and due to the chamber drag are responsible for the high resulting fuel consumption.

After this description of the more fundamental relation a number of detailed problems connected with the pressure-jet drive with tip burning are considered.

## Head and Blades

The higher compression ratios used for the tip burning case allow a more compact head design. On the other hand, the rotating seals need careful selection of sealing materials. Carbon seals running on case hardened steel for the main rotating seals and metal-to-metal seals in conjunction with piston rings for the feathering and flapping hinges, proved satisfactory. The high air temperature calls for careful design and arrangement of the tip-jet fuel system inside the head in order

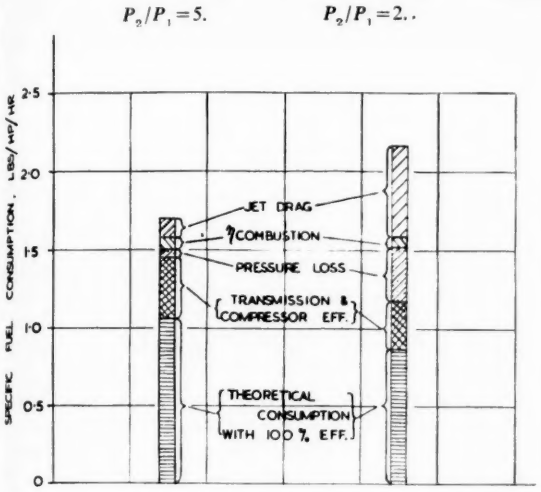



FIGURE 4. Comparison of specific fuel consumption of a pressure jet.

to prevent vaporisation. The elimination of the torque at the blade root of the shaft-driven rotor system allows the drag hinge to be dispensed with, which contributes to the reduced vibration levels of the jet-driven rotor. The universal joint suspended head where the differential blade flapping with respect to the axis of rotation takes place only during control movements of the disc path plane, proved very successful and is used in a number of flying pressure-jet helicopters. Fig. 5 shows a number of blade designs used in pressure-jet helicopters. A limit for the moment of area within the blade contour which may be used as a duct is the chordwise position of the centre of gravity which, for normal aerofoils should be kept between 22 and 25 per cent.

The necessary duct area ratio decreases with the size of the rotor blade and the larger the helicopter, the easier becomes the blade design from this point of view. The Djinn blade and the Ultra Light blade are typical blade conceptions for a small pressure-jet helicopter where the spar forms the duct and the combined stresses of centrifugal force and inner pressure call for tapered wall thickness with blade radius.

In a big blade such as the Rotodyne blade, it is possible to accommodate the necessary duct area in round tubes of small wall thickness inside the blade structure. These tubes carry only their own centrifugal force, while the blade and chamber centrifugal force is carried by a separate spar arrangement. The differential expansion between the blade structure and the air duct due to temperature difference is taken care of by a sleeve arrangement at the outer end of the blade structure where the jet units are connected.

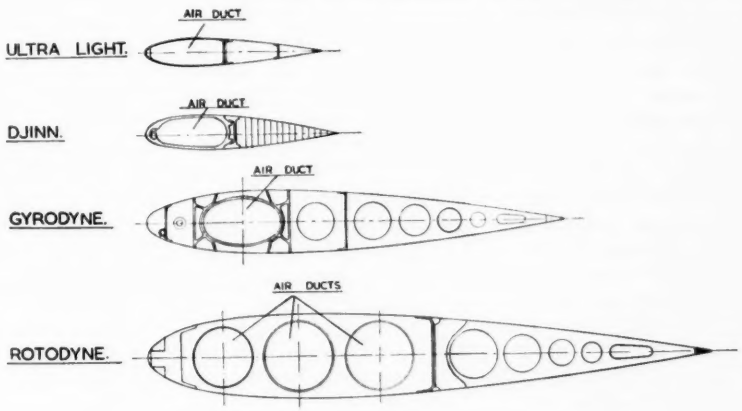



FIGURE 5. Blade ducts for pressure jets.

Keeping the duct area in a constant ratio to the entire cross-sectional area, the pressure drop changes approximately inversely proportional to the fifth power of the blade chord. The efficient performance and structural reliability of a jet rotor system depends a great deal on the correct compromise between the aerodynamics of the rotor as a lifting device and the duct aerodynamics and structural limitations of the rotor as a part of the propulsion system. Careful design studies are necessary before a complete design for a jet rotor system can be decided upon. Assuming that the compression ratio is fixed, a design study based on a structural duct area ratio can be composed with the solidity, tip speed and disc load as parameters. There is an optimum solidity at a certain tip speed regarding specific fuel consumption; this solidity may not be acceptable from the aircraft's aerodynamic point of view, but indicates in which direction change in duct design is necessary.

## Fuel System

In a helicopter employing pressure-jet drive, fuel to the tip-jet units is usually supplied by a pump in the fuselage by way of a regulator to the rotor head and thence along each blade. Because of the high centrifugal force on any device installed at the blade tip, the use of any valve or regulator near the combustion chamber is usually avoided. It is conceivable that fuel regulation could be achieved by varying the pump delivery pressure proportionally to the square of the flow required. However, for such a scheme to be effective the fuel pipes in the blades would have to run full and pressures up to about 3,000 lb./in.² would be reached at the sprayer due solely to centrifugal action of the rotor. The minimum flow would have to be achieved with this high pressure and the pump would then have to supply the excess pressure required to reach the maximum flow. As this flow is about four times the minimum, a pump pressure for the full flow of 48,000 lb./in.² would be required. This is considered impracticable, as is the sprayer for the pressure range because of the fine tolerances involved and the high degree of fuel cleanliness required.

Thus for a hub metering system, the fuel flow downstream of the regulator is influenced by only two

devices, a distributor or restrictor in the head which ensures equal division of fuel to the blades and, the fuel sprayer in the combustion chamber which breaks up the fuel into droplets for ease of mixing with air and subsequent combustion.

The sprayer breaks the fuel into fine droplets by producing a high level of turbulence and large shearing forces in the liquid film leaving the device. In combustion chambers of round section a swirling flow produces the desired effect, so the fuel is discharged by way of a vortex chamber to give a hollow conical envelope of fuel. In chambers of elliptic or rectangular section an asymmetrical spray pattern is desirable and the fuel is discharged through opposing jets so that impingement of the fuel streams produces a fan-like film. Whatever the final pattern, the flow of fuel is proportioned to the square root of the fuel pressure at the sprayer and, in gas turbine nomenclature, the sprayer characteristic is defined as

$$\text{Flow number} = \frac{\text{Fuel flow in gallons/hour}}{\sqrt{(\text{Fuel pressure in excess of surrounding air at sprayer in lb./in.}^2)}}$$

After leaving the restrictor the fuel flows outwards along the blade fuel pipe, the flow being aided by the pumping effect of the rotor rotation and retarded by pipe friction. If pipe friction is high, as in a small bore pipe, or one with a rough wall, the pressure in the fuel line near the hub may fall faster with radius than can be compensated by the rotor pumping effect. For a given pump pressure or, more correctly, restrictor outlet pressure, there will be a maximum fuel flow determined by the formation of vapour in the fuel pipe which breaks the liquid column at some radius near the hub. This critical flow cannot be increased without an increase of pressure at the restrictor outlet because pressure at the "break" will be maintained at the fuel vapour pressure. This implies that there is a minimum pipe size if fuel flow, restrictor outlet pressure and rotor r.p.m. are fixed.

If the minimum pipe size is exceeded, fuel continues to flow outwards under ever-increasing pumping force until it is possible for the fuel to accelerate, causing cavitation, the fuel flowing in a segment on the trailing side of the tube as a result of Coriolis forces. Immediately upstream of the sprayer a fuel column will build up until the pressure at the sprayer enables the flow out on to the combustion chamber to balance the flow of fuel arriving at the top of the column. The column height will be that which gives sufficient pressure to overcome pipe friction in the column and meet the requirement of the sprayer. For a particular rotor r.p.m., sprayer flow number and pipe size, every fuel flow rate will have an appropriate column height.

When the fuel flow is suddenly altered by the pilot this change occurs in the fuselage, usually at the fuel regulator and, with negligible delay, is relayed to the restrictor. Fuel leaving the restrictor flows along the blade, as already described, until it reaches the top of the column which now assumes a new height. These two stages, namely, the traversing of the blade and the filling (or emptying) of the pipe from one column height to another, constitute the delay in fuel response likely to be experienced in all pressure-jet helicopters where fuel is controlled within the fuselage rather than at the tip-jet unit.

## Effect of Pipe Size

### GENERAL CONSIDERATION

With the restrictor placed close to the hub the fuel flow can only be influenced by rotor speed, blade fuel pipe size and sprayer flow member. A large pipe will not run full and is capable of a high mass flow with low friction, but the amount of fuel involved in a change of column height will be large. Alternatively, the small fuel pipe, while requiring only small volumes of fuel to effect column changes, incurs higher frictional resistance to flow so that, as has been mentioned, there is a minimum size of pipe capable of sustaining a particular flow near the hub. As the pumping effect of the rotor increases with increasing rotor radius, progressively smaller pipe diameters are possible, provided that, as will be seen later, the pressure at the sprayer at the minimum fuel flow does not fall below about 50 lb./in.$^2$, which is the minimum pressure required for satisfactory "atomisation."

From these considerations, two broad limits affecting pipe size and flow number are recognisable. At the minimum rotor speed the minimum fuel flow must give a pressure of 50 lb./in.$^2$ at the sprayer, while the maximum fuel flow at this low rotor r.p.m. should be just sufficient to fill the fuel pipe from sprayer to restrictor.

## Combustion Chambers

With the pressure-jet helicopter where fuel is burnt at the rotor blade tips, the combustion chambers may differ from normal gas turbine practice for two main reasons: —

(*i*) Due to the absence of a turbine downstream of the chamber, both higher outlet temperatures and greater transverse variation in these temperatures are permissible.

(*ii*) Because of its location, it is particularly important to keep the size of the chamber to a minimum consistent with other requirements.

Two approaches to the problem of chamber size are possible. Firstly, the position of the chamber may be varied, the alternatives being (*a*) a radial chamber built into the aerofoil section of the blade, (*b*) a tangential chamber attached to the end of the blade.

The former produces a minimum of parasitic drag, the only protuberance into the external air flow being the nozzle. Unfortunately, an unsatisfactorily shaped combustion space is necessitated and it entails a loss in turning the heated high speed gas through 90°. If a radial chamber is selected, the only parameter which the designer can vary is the length.

The tangential chamber involves asymmetric air entry with consequent pressure drop, but this may penalise the burning conditions less than the awkward chamber cross section of the radial version. It also allows a compromise between chamber volume and combustion considerations. Furthermore, its end plate effect decreases the aerodynamic tip loss of the rotor blade.

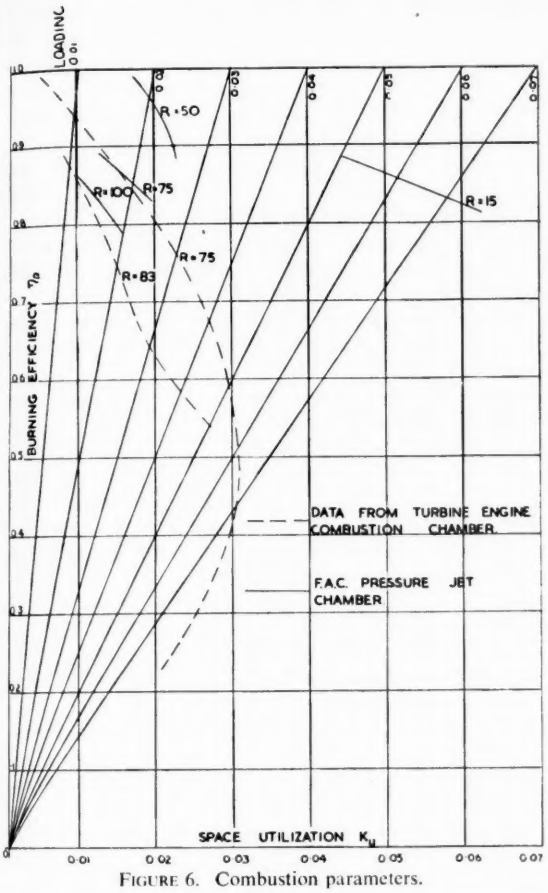



FIGURE 6. Combustion parameters.

The utilisation of combustion volume for producing energy may be varied by altering the air-fuel ratio. To consider this effect it is necessary to outline a non-dimensional method of correlating the relevant combustion parameters.

Dimensional analysis suggests that for a given air-fuel ratio the burning efficiency is a function of the loading:

$$\eta_B = f(L) \quad . \quad . \quad (1)$$

$$\text{where } L = \frac{\text{Heat energy in fuel supply}}{\text{Max. possible fuel release per unit vol.} \times \text{chamber vol.}} . \quad (2)$$

and $\eta_B$ = burning efficiency.

For any particular chamber the form of this function depends on the pressure drop across the chamber and the success with which it is used by the flame tube in producing satisfactory mixing.

The volume rate of heat release in the combustion space is represented by the utilisation factor

$$Ku = \frac{\text{Actual heat release}}{\text{Max. possible heat release per ft.}^3 \times \text{chamber vol.}} \quad . \quad . \quad . \quad (3)$$

Since actual heat release $= \eta_B \times$ heat energy in fuel supply, it follows from equations (2) and (3) that

$$Ku = \eta_B L \quad . \quad . \quad . \quad (4)$$

These three variables $\eta_B$, $L$ and $Ku$ may be presented on a graph as shown in Fig. 6 where test results of $\eta_B$ are plotted against $L$ and in practice give single lines for each air-fuel ratio. Data for it was taken from the Bragg and Holliday paper presented at the Second AGARD Combustion Colloquium and from tests of Fairey Aviation Co. weak and rich mixture chambers (*see* Ref. 4). From it, it may be seen that at constant burning efficiency, rich mixtures allow higher loading and, consequently, better utilisation.

An example may show the effect on combustion volume of a constant thrust chamber of varying the air-fuel ratio from 75, a normal value for gas turbines, to 15, the stoichiometric value.

Assumptions:

Thrust = 100 lb. Compression ratio = 4·1

$\eta_B$ = 85 per cent = constant nozzle efficiency of 95 per cent

The air mass flows $M_A$ required are:

1·36 lb./sec. for $R = 75$ and 0·855 lb./sec. for $R = 15$.

From Fig. 6 the $Ku$ which can be achieved in practice may be taken from Fairey Aviation Company chambers.

At $R = 75$ $\quad Ku = 0{\cdot}0175$

$R = 15$ $\quad Ku = 0{\cdot}053$

Replacing the term, heat energy in fuel supply, in equation (2) by $\frac{M_A}{R+1} \times$ calorific value of fuel, it follows from equation (3) since $\eta_B$ = constant

$$\text{that} \quad \frac{Ku\,75}{Ku\,15} = \frac{V\,15}{V\,75} \times \frac{M\,75}{M\,15} \times \frac{16}{76}$$

which gives the ratio of the volumes

$$\frac{V\,15}{V\,75} = \frac{Ku\,75}{Ku\,15} \times \frac{M_A 15}{M_A 75} \times \frac{76}{16} = 0{\cdot}98.$$

Hence it appears that although a smaller volume might be expected for richer mixtures, because of the higher space utilisation, no significant decrease materialised. The reason for this is that a given thrust requires a fixed amount of energy. The energy may come either in

(*i*) high mass flow at low temperature

(*ii*) low mass flow at high temperature.

In the latter case, a greater heat release is needed in the chamber. Therefore, although the air mass flow is less and the space utilisation greater, there is not much saving in volume. For a combustion chamber used as a tip propulsion unit it is more realistic to compare the combustion performance at constant net thrust, instead of constant gross thrust as above. For this example at constant net thrust and an assumed tip speed of 650 ft./sec. the volume ratio decreases to 0·85, which is appreciably favourable to the rich mixture case. The smaller combustion chambers in the rich mixture case improves the aerodynamics of the rotor which, combined with the lower necessary installed engine weight (due to the lower mass/flow requirements), shows the advantage of the rich mixture case.

Another special condition applying only to tip-jet chambers is the strong force field in which they operate due to centrifugal acceleration from rotation. This presents special structural problems. It may also affect

combustion by influencing the paths of fuel particles. Here high pressure atomisation giving small droplet size is an advantage in increasing the ratio of air drag to inertia effects.

## The Measuring of the Thrust on the Rotating Chamber

The total thrust of the rotating chamber can be lower than the static thrust due to less efficient burning in the rotating chamber. The centrifugal force acting on the fuel spray, as well as on the air flow, may justify such an assumption. Due to the momentum loss any percentage of total thrust lost results in a higher percentage loss of net thrust and is therefore very critical for the correct rotor power evaluation. As it is practically impossible to measure the total thrust of the rotating chamber directly, another parameter was used to establish the thrust while rotating. It could be shown that the thrust of a choked jet of a constant orifice is a function of the total inner pressure only and is independent of chamber temperature and mass flow. This relation in this simple form is only correct for limited changes of chamber temperature, where the ratio of the specific heat and the discharge coefficient can be considered as constant.

From this relation it can be concluded that the thrust of the rotating chamber can be determined by measuring the inner pressure of the chamber at a point previously used for establishing the static characteristic and by measuring the mass flow and fuel flow. The pressure should be taken at a point which is unaffected by changing of the burning pattern, or changing of the flow condition and such a point is easily found in a number of static tests. The measuring of the air and fuel flow delivered to the rotating chamber presents no problem and can be done with standard equipment. To measure the inner pressure, the pressure point on the chamber is connected by way of a capillary tube inside the blade to a pressure transmitter on top of the rotating part of the head. The pressure signals are transmitted electrically by way of slip rings to the stationary part of the head and, from there, to an indicating instrument in the test cabin. The temperature of the stationary air column inside the capillary tube is measured by thermocouples, the indications of which are also transmitted electrically by slip rings to a temperature indicator in the test cabin. The actual pressure in the combustion chamber has to be computed from the indicated pressure and the pumping pressure of the stationary air column which, in turn, can be calculated from the tip speed and air column temperature readings.

From this calculated pressure of the rotating chamber the thrust can be established, using its static characteristic. By comparing the measured mass flow and fuel flow on the rotating chamber with those of the static test characteristic at the same combustion chamber pressure, a possible change of burning efficiency in the rotating chamber can be detected.

## Noise

This paper would be incomplete without some discussion of the noise problem. This is the most criticised feature of all tip-jet drives and the objection of the operator who intends to use the rotorcraft for passenger transport right into city centres is fully understandable. I had not intended to discuss noise reduction devices which would fall outside the scope of the paper, but the correct assumption of the noise level of a jet rotor system is a new feature among design studies and therefore should be included in a general discussion of design and performance problems.

Wide and intensive research into the noise and its suppression is now conducted in many places and this is still a problem full of contradicting expressions.

The following conclusions should, therefore, only be considered as based on tests made by the Fairey Aviation Company.

There appears to be a distinct difference in the laws governing the cold air jet and the hot and highly turbulent jet from a pressure-jet combustion chamber.

The noise level of a cold air jet can be concluded from the Lighthill theory as being proportional to the eighth power of the jet speed and first power of the nozzle area. A great number of noise investigations with cold air jets confirm this relation with only slight differences in the values of the speed exponent. We found, by using very high turbulent cold air jets, that up to choking pressure ratios noise increases as predicted with the eighth or ninth power of the speed, then comes a range between pressure ratios of 1·8 and 2·5 where the noise tends to increase with the thirteenth power. Above pressure ratios of 2·5 it appears that the noise follows again the eighth power rule. As soon as burning is introduced, the law changes rapidly. Table I shows test results with an unsilenced round combustion chamber.

TABLE I

| *Air-Fuel ratio* | *Thrust lb.* | *Chamber pressure lb./in.²* | *Mass Flow lb./sec.* |
|---|---|---|---|
| ∞ | Constant | Constant | 100 per cent |
| 76 | Constant | Constant | 67 per cent |
| 17 | Constant | Constant | 44 per cent |

| *Jet Speed ft./sec.* | *Noise output. Watts* | *Noise level at 100 ft. db.* | *Noise level at 100 ft. based on $V^8$ law. db.* |
|---|---|---|---|
| 1,850 | 284 | 106·7 | 106·7 |
| 2,560 | 650 | 110·5 | 116·7 |
| 4,110 | 1,228 | 113·2 | 133·0 |

From these results, which were taken from noise spectrum measurements, on a radius of 100 ft. and at stations covering the full 360° round the nozzle, it can be concluded that the noise level of the burning jet increases much less with speed than for a cold air jet and there is a falling power with jet temperature. Fig. 7 shows a plot of the power by which the noise increases with speed when increasing the fuel-air ratio from infinity (cold air jet) to a certain value. This is a most valuable relation for assessing the noise of a burning jet from a cold air model test.

A better agreement with the Lighthill speed exponent can be obtained by introducing the density of the jet efflux into the Lighthill parameter. The speed exponent

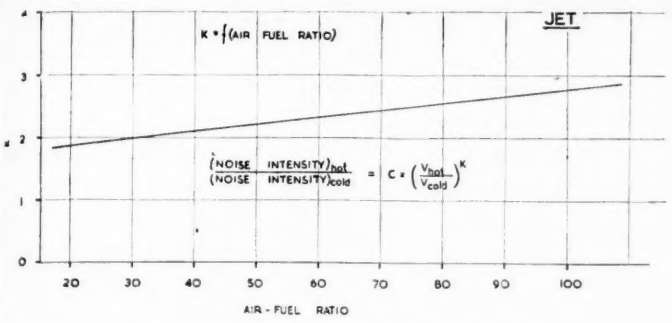



FIGURE 7. Speed law exponent for burning pressure-jet.

of approximately 2·5 as given in Fig. 7 will then rise to 6·6.

On the other hand, replacing the Laval outlet speed by the speed of sound in the jet throat, the speed exponent is reduced again to a value around 2·5.

Further tests with rotating cold and burning jets indicate that the $V^8$ law can be applied in order to assess the noise reduction due to the lower differential speed between the jet and the surrounding air with the former rotating. The area law and the distance law (*i.e.* reduction of noise intensity with the square of the distance) was also found to be correct inside acceptable limits.

Parallel to these investigations about principal relationships an intensive noise reduction research programme was carried out, which achieved noise reduction of over 13 db. on burning jets and there are definite indications that even higher noise reduction should be possible. According to Professor Richards[5] there are even claims of possible noise reductions up to 20 db.

With these findings and established laws, the ratio of the noise output of a full scale rotating burning jet to the noise output of a static cold air model jet is proportional to:

(*a*) The ratio of Laval outlet speeds to the power of $K$ ($K$ being a function of fuel-air ratio, *see* Fig. 7).
(*b*) The ratio of the outlet areas.
(*c*) The ratio of the differential speed between nozzle outlet speed and tip speed to the nozzle outlet speed to the eighth power.

For example: the noise output of a cold air model jet is measured to be 284 Watts. This corresponds to a noise level of 106 db. at 100 ft. distance. From this, the noise level of a full scale rotating jet with 6·8 times the thrust output, working under the same pressure but at an air-fuel ratio of 17 and rotating at a tip speed of 720 ft./sec. should be predicted. With the foregoing relations the noise output of the full scale jet is calculated to be 1,810 Watts corresponding to 114·9 db. at 100 ft. distance. Assuming an achievable noise reduction of 13 db. the noise level of the silenced full scale jet would be 101·9 db., which is reduced to 96 db. at 200 ft. distance.

Finally, noise reduction by producing the equivalent rotor power by a cold air jet of the same pressure ratio can be calculated with the figures in Table I and the foregoing relation.

The noise level, applying the same silencing device, would be: —

db. (cold air jet) = 99·8 − 13 = 86·8 db.

The reduction in noise is 15 db.

The engine power required would be three times the engine power for the tip burning case and the blade and head design has to be modified to accommodate three times the ducting space.

## Conclusions

After this general review and detailed treatment of some of the special problems of the pressure-jet helicopter, I would like to summarise the conclusions.

Each configuration of tip-jet drive has applications for which it alone is superior to the rest; I feel that the two extreme solutions give the best demonstration for the limits of adaptability. The ram-jet rotor will always be attractive for all cases where utmost simplicity and low initial and maintenance costs are the deciding factor, while endurance, aerodynamics in forward flight and economy are of secondary consideration.

On the other hand, I am of the opinion that, for the large single rotor helicopter for medium distance intercity transport, with short hovering times and with optimum aerodynamic requirements for fast cruising speed with unloaded and wind milling rotor, the pressure-jet with tip burning is the most rewarding solution.

The main objection against all tip-jet rotor systems, the high noise level, is fully recognised, but results of intensive noise reduction research are encouraging and I firmly believe that this problem will be solved satisfactorily.

ACKNOWLEDGMENT

I wish to thank the Fairey Aviation Co. Ltd. for help in the preparation of, and permission to publish, this paper.

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# TECHNICAL NOTES

## *A Method for Calculating the Energy Available in the Exhaust Gas at the Inlet End of an Exhaust Pipe of a Two- or Four-Stroke Cycle Engine*

by

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THE NATURE of the flow in the exhaust pipe of a turbo-charged two- or four-stroke cycle engine is in general so complex that practical tests on running machinery are necessary to compare the merits of one exhaust system with another. The analysis of the results of these tests is rendered somewhat difficult by the lack of reliable methods for measuring accurately the transient temperatures in the exhaust pipe. Indicator diagrams have therefore to be used in conjunction with empirical methods for computing the exhaust energy, these methods being associated with the measured exhaust pyrometer temperature. In calculating the theoretical possibilities of different exhaust pipe configurations for four-stroke cycle engines Jenny[1, 2] has suggested that the total energy was the sum of the internal and kinetic energies of the gas entering the pipe. In the present treatment it is suggested that a better criterion is the enthalpy available if the gas expands isentropically from stagnation conditions to atmospheric pressure, the justification being that this would be the energy utilised in the turbine.

A method is given for computing the energy available in the exhaust gas at the inlet end of the pipe assuming quasi-steady flow across the ports, adiabatic gas exchange or flow, and in certain cases isentropic expansion. Using these assumptions the only other test data required are indicator diagrams in the inlet manifold, in the cylinder and in the exhaust pipe adjacent to the cylinder. The method is advanced as the basis of an analytical technique which may require modification in the light of experience. Since the theory is associated only with the gas entering the pipe the calculated energy available will be greater than the actual energy available in the gas entering the turbine, although this may not be so for engines with short pipes. Nevertheless, for exhaust systems with identical configurations such factors as exhaust valve or port design or other physical variables at the inlet end of the pipe may be investigated by this method.

### NOTATION

- $a$ speed of sound (ft./sec.)
- $m$ mass flow rate (slugs/sec.)
- $n$ connecting rod length : crank ratio
- $p$ pressure (lb./ft.$^2$)
- $s$ entropy (ft. lb./slug per ° Rankine)
- $t$ time (secs.)
- $u$ velocity (ft./sec.)
- $E$ instantaneous energy (ft. lb./sec.)
- $F$ area (ft.$^2$)
- $H$ total heat or enthalpy drop (ft. lb./slug)
- $L$ cylinder length (ft.)
- $N$ engine speed (r.p.m.)
- $T$ temperature (° Rankine)
- $V$ volume (ft.$^3$)
- $W$ mass (slug)
- $\rho$ density (slugs/ft.$^3$)
- $\gamma$ ratio of specific heats
- $\alpha$ crank angle (degrees)
- $c_p$ specific heat at constant pressure (ft. lb./slug/° Rankine)
- $c_v$ specific heat at constant volume (ft. lb./slug/° Rankine)

*Suffixes*

- i inlet conditions or port, stagnation condition in inlet manifold
- 0 stagnation condition in cylinder
- A atmospheric pressure
- 1 cylinder
- 2 *vena contracta* of exhaust port or valves
- 3 exhaust pipe inlet end

*Non-dimensional parameters*

$$\xi = \frac{m_3 a_0}{p_0 F_3}$$ non-dimensional mass flow

$$\lambda = \frac{H}{a_0^2}$$ non-dimensional enthalpy drop

$$\varepsilon = \frac{E}{F_3 p_0 a_0}$$ non-dimensional instantaneous energy

$$\pi = \frac{p_3}{p_0}$$

$$r = \frac{p_0}{p_A}$$

$$x = \frac{p_0}{p_1}$$

$$\psi = \gamma\left(\frac{2}{\gamma-1}\right)^{\frac{1}{2}} x^{(1-\gamma)/\gamma}\left(1 - x^{(\gamma-1)/\gamma}\right)^{\frac{1}{2}}$$

$$\Phi = \psi x$$

$$U = \frac{u_3}{a_0}$$

$$\nu = \frac{u_3/a_0}{1 - \left(\frac{\gamma-1}{2}\right)\left(\frac{u_3}{a_0}\right)^2}$$

The gas exchange process can be sub-divided into two phases, the exhaust blowdown and the scavenge and charging period. During the exhaust blowdown period the expansion will be isentropic, while during the scavenge and charging if heat transfer is neglected to and from the cylinder walls the process will be adiabatic but not isentropic. The expansion through the inlet ports and the exhaust ports will be assumed to be isentropic, but the

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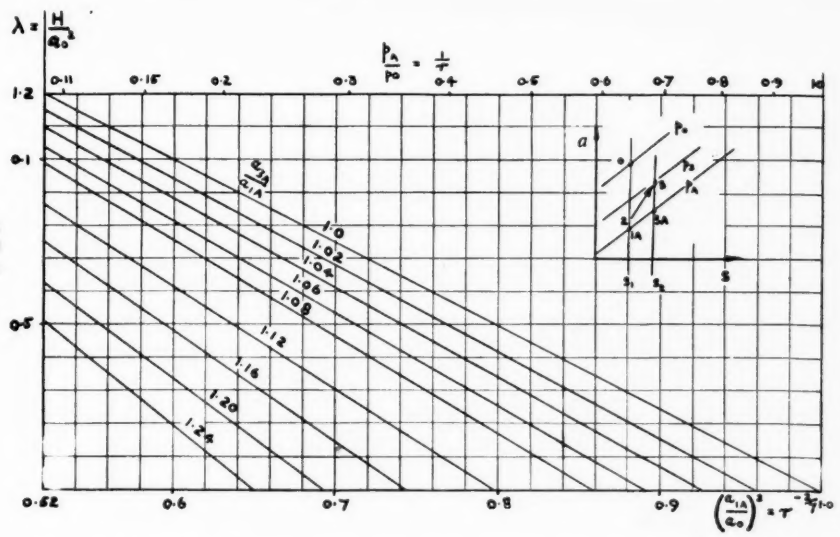


FIGURE 1. Relationship between dimensionless enthalpy drop and (pressure ratio)$^{-2/7}$ in cylinder for constant entropy changes $\frac{a_{3A}}{a_{1A}}$

pressure recovery on the downstream side will be assumed to be adiabatic.

The flow through the exhaust ports into the pipe can be represented on a speed of sound-entropy diagram (*a-s* diagram) (Fig. 1). The state of the gas in the cylinder is at 0, in the exhaust port at 2 and in the pipe at 3. The gas expands isentropically from stagnation pressure $p_0$ to pressure $p_2$ in the port and then enters the exhaust pipe with pressure rise $p_2$ to $p_3$. If $a_{1A}$ is the speed of sound at atmospheric pressure on the cylinder isentrope $s_1$ and $a_{3A}$ is the corresponding speed of sound on the pipe isentrope $s_3$ then

$$\frac{a_{3A}}{a_{1A}} = \exp\frac{s_3 - s_1}{2c_p}.$$

The term $\frac{a_{3A}}{a_{1A}}$ will be called the entropy change.

We shall consider the total energy available per unit mass as the total kinetic energy if the gas were expanded isentropically from stagnation conditions to atmospheric pressure, that is, the change in total heat (or enthalpy) between stagnation conditions and atmospheric pressure. If $H$ is the change in specific enthalpy in ft. lb./slug then from the theorem of conservation of energy for quasi-steady flow across the ports:—

$$\frac{a_0^2}{\gamma-1} = \frac{a_3^2}{\gamma-1} + \frac{u_3^2}{2} = H + \frac{a_{3A}^2}{\gamma-1}$$

$$H = \frac{1}{\gamma-1}[a_0^2 - a_{3A}^2] \text{ ft. lb./slug.}$$

Let $\lambda = \frac{H}{a_0^2}$ called non-dimensional enthalpy drop

then

$$\left.\begin{aligned}\lambda &= \frac{1}{\gamma-1}\left[1 - \left(\frac{a_{3A}}{a_{1A}}\right)^2\left(\frac{a_{1A}}{a_0}\right)^2\right] \\ \lambda &= \frac{1}{\gamma-1}\left[1 - \left(\frac{a_{3A}}{a_{1A}}\right)^2\left(\frac{p_A}{p_0}\right)^{\frac{\gamma-1}{\gamma}}\right]\end{aligned}\right\} \quad . \quad (1)$$

A series of curves for constant $\frac{a_{3A}}{a_{1A}}$ for $\gamma = 1{\cdot}4$ are given in Fig. 1. By plotting $\lambda$ against $\left(\frac{a_{1A}}{a_0}\right)^2$ these will become straight lines.

The mass flow entering the pipe for quasi-steady flow will be given by

$$m_3 = \rho_3 u_3 F_3 \text{ slugs/second.}$$

In non-dimensional form

$$\xi = \frac{m_3 a_0}{F_3 p_0} = \frac{\gamma\left(\frac{p_3}{p_0}\right)\left(\frac{u_3}{a_0}\right)}{1 - \left(\frac{\gamma-1}{2}\right)\left(\frac{u_3}{a_0}\right)^2} \quad . \quad . \quad (2)$$

$\xi$ = non-dimensional mass flow.

Let $\nu = \dfrac{u_3/a_0}{1 - \left(\frac{\gamma-1}{2}\right)\left(\frac{u_3}{a_0}\right)^2} = \dfrac{U}{1 - \left(\frac{\gamma-1}{2}\right)U^2}$

$$\pi = \frac{p_3}{p_0}$$

then $$\xi = \gamma\nu\pi \quad . \quad . \quad . \quad (3)$$

The entropy change $\frac{a_{3A}}{a_{1A}}$ is related to $U$ and $\pi$:—

$$\frac{a_{3A}}{a_{1A}} = \left(1 - \frac{\gamma-1}{2}U^2\right)^{\frac{1}{2}}\pi^{(1-\gamma)/2\gamma}. \quad . \quad (4)$$

It is convenient to express

$$\xi = f_1\left(\frac{a_{3A}}{a_{1A}}, \pi\right)$$

This is best carried out by plotting equation (2) or (3), $\xi = f_2(U, \pi)$ and equation (4), $\frac{a_{3A}}{a_{1A}} = f_3(\pi, U)$ on the same graph with $U$, $\pi$ co-ordinates; intersections of these curves will give the numerical solution of $\xi = f_1\left(\frac{a_{3A}}{a_{1A}}, \pi\right)$. These may be rapidly computed for $\gamma = 1{\cdot}4$ by means of the usual compressible flow tables [3, 4]. The graphs in Fig. 2 show $\xi$ against $\pi$ for lines of constant $\left(\frac{a_{3A}}{a_{1A}}\right)$ ($\gamma = 1{\cdot}4$).

The instantaneous energy in ft. lb./second, $E$, will be given by

$$E = m_3 H = \lambda\xi F_3 p_0 a_0 \text{ ft. lb./sec.}$$

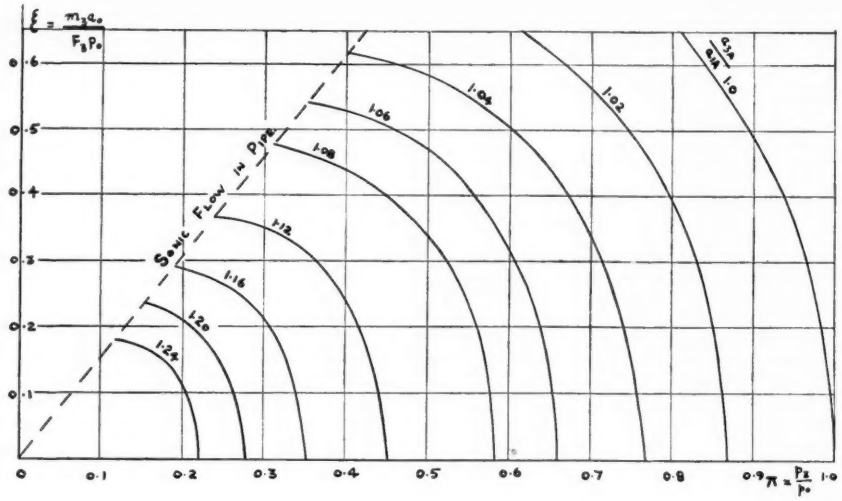


FIGURE 2. Relationship between dimensionless mass flow and pressure ratio $p_3/p_1$ for constant entropy changes $\frac{a_{3A}}{a_{1A}}$

and in non-dimensional form

$$\varepsilon = \frac{E}{F_3 p_0 a_0} = \lambda\xi.$$

During the gas exchange process $\lambda$ and $\xi$ will vary continuously; the total work available per cycle is therefore:—

$$\int E dt = \frac{F_3}{6N} \int \varepsilon p_0 a_0 d\alpha \text{ (ft. lb.)}$$

$N$ = engine speed in r.p.m.
and the corresponding horsepower:—

$$\frac{F_3}{198{,}000} \int \varepsilon p_0 a_0 d\alpha. \quad . \quad . \quad . \quad (5)$$

APPLICATION TO ENGINE

To apply the above theory to the engine it is necessary to calculate the mass flow $\xi$ whence, from the pressures in the pipe and cylinder, the entropy change can be calculated from Fig. 2, and the enthalpy $\lambda$ from Fig. 1.

The mass change in the cylinder is given by

$$\frac{dW}{W} = \frac{dp_0}{p_0} + \frac{dV}{V} - \frac{dT_0}{T_0}.$$

The energy balance

$$dW_1 (c_p)_1 T_1 - dW_3 c_p T_0 = W c_v dT_0 + dW c_v T_0 + p_0 dV.$$

The suffix "i" refers to inlet air. If the specific heats are assumed constant $(c_p)_1 = c_p$ then by combining the above two equations and differentiating we have

$$m_3 a_0^2 = m_1 a_1^2 - V \frac{dp_0}{dt} - \gamma p_0 \frac{dV}{dt} \quad . \quad . \quad (6)$$

where $m_1$ = inlet mass flow slug per second
$a_1$ = stagnation speed of sound in inlet manifold.

In non-dimensional form equation (6) becomes

$$\xi = \frac{m_3 a_0}{F_3 p_0}$$

$$= \left(\frac{a_1}{a_0}\right)\left(\frac{F_1}{F_3}\right)\psi - \frac{6N}{a_0}\left(\frac{F_1}{F_3}\right)\left[\gamma \frac{dL}{d\alpha} + \frac{L}{r}\frac{dr}{d\alpha}\right]$$

$p_1$ inlet air pressure
$p_0$ cylinder pressure
$F_1$ effective area of inlet valve or port
$F_1$ cylinder cross sectional area
$F_3$ exhaust pipe cross sectional area
$r$ non-dimensional cylinder pressure $= \frac{p_0}{p_A}$
$N$ r.p.m.
$L$ cylinder length (ft.)
$\psi = \gamma\left(\frac{2}{\gamma-1}\right)^{\frac{1}{2}} x^{-\frac{1-\gamma}{\gamma}} \left(1 - x^{-\frac{1-\gamma}{\gamma}}\right)^{\frac{1}{2}}$ for subsonic flow through inlet valve or port (*see* Fig. 3 for $\gamma = 1{\cdot}4$)
$x = p_0/p_1$
$dL$ rate of change of cylinder length per crank angle degree
$d\alpha$ $= \frac{s}{2}\left[\sin\alpha + \frac{1}{2n}\sin 2\alpha\right]\frac{\pi}{180}$
$n$ connecting rod length : crank ratio
$s$ stroke (strokes if opposed piston engine) in feet

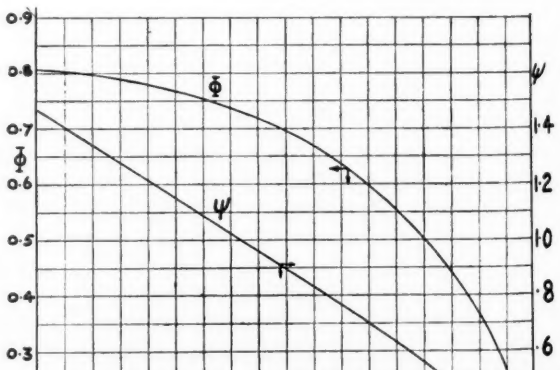


FIGURE 3. Inlet flow functions $\phi$ and $\psi$.
Inlet mass flow $m_1 = \frac{p_1 F_1}{a_1} \Phi$ slugs/sec.

$\frac{dr}{d\alpha}$ rate of change of non-dimensional cylinder pressure obtained directly from the indicator diagram

$a_0$ stagnation speed of sound in cylinder

When the inlet ports are closed $F_1=0$.

$$\xi = -\frac{6N}{a_0}\left(\frac{F_1}{F_3}\right)\left[\gamma\ \frac{dL}{d\alpha} + \frac{L}{r}\frac{dr}{d\alpha}\right].$$

This corresponds to exhaust blowdown, $a_0$ can be calculated by assuming isentropic expansion from release temperature and pressure.

When the inlet ports are open the full expression for $\xi$ is used, $a_0$ has now to be calculated by an iterative method. One method is to take equal crank angle increments $\Delta\alpha$ and continuously calculate $a_0$. The procedure would be to estimate $a_0$ and calculate $\xi$ and thence obtain $m_3$. The inlet mass flow $m_1 = \frac{p_1 F_1 \Phi}{a_1}$ where $\Phi = \psi x$. (*See* Fig. 3 for $\gamma = 1{\cdot}4$) whence the total change in mass can be obtained.

$$dW = (m_1 - m_3)\frac{\Delta\alpha}{6N}$$

and $a_0$ recalculated

$$a_0^2 = \frac{\gamma p_0 V}{W + dW}.$$

Where $p_0$ and $V$ are the mean pressure and volume for the interval $\Delta\alpha$.

The new value of $a_0$ is then compared with the estimated value.

## Conclusion

A method is given for calculating the energy available in the exhaust gas as it enters the exhaust pipe. The method defines the energy as that which would be available if the gas expanded to atmospheric pressure from stagnation conditions. By assuming adiabatic flow and gas exchange non-dimensional parameters are developed enabling the energy to be calculated from the pressures obtained by the usual indicating techniques.

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# *An Experiment on Flat Plate Turbulent Boundary Layer Flow—The Effect of Local Fluid Addition on Friction and Velocity Distribution*


by

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FOR A FLAT SURFACE at zero incidence in a deep uniform stream, the total frictional drag $F$ per unit width of the surface can be calculated from the well known equation

$$F = \rho \int_0^{\delta} u\,(u_0 - u)\,dy \quad . \quad . \quad . \quad (1)$$

Using the momentum principle in the usual way, it can be shown that when fluid is added locally to the flow through the surface, the total frictional drag $F_1$ is given by

$$F_1 = \rho \int_0^{\delta_1} u\,(u_0 - u)\,dy - wu_0 + wu_1 \cos\theta \quad . \quad (2)$$

where $\theta$ is the angle between the flat surface on the downstream side and the direction of the fluid addition.

Equation (2) is quite general, and is derived without any reference to the position at which the fluid is added to the flow. Once the velocity distribution is known at any section $x\,(>x_1)$, the frictional drag for that length may be determined.

## Experimental Work

Figure 1 shows the essentials of the experimental rig used to determine the effect on the flow characteristics, friction and velocity distribution, of adding air to a fully developed turbulent boundary layer flow on a flate plate. The secondary air was admitted by a rectangular orifice of high aspect ratio, in which case the flow was considered to be two-dimensional. To observe the penetration effects of the secondary air it was added perpendicularly to the

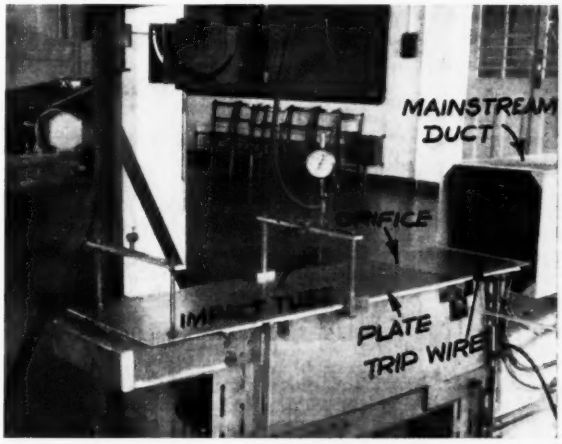


Figure 1. Plate assembly.

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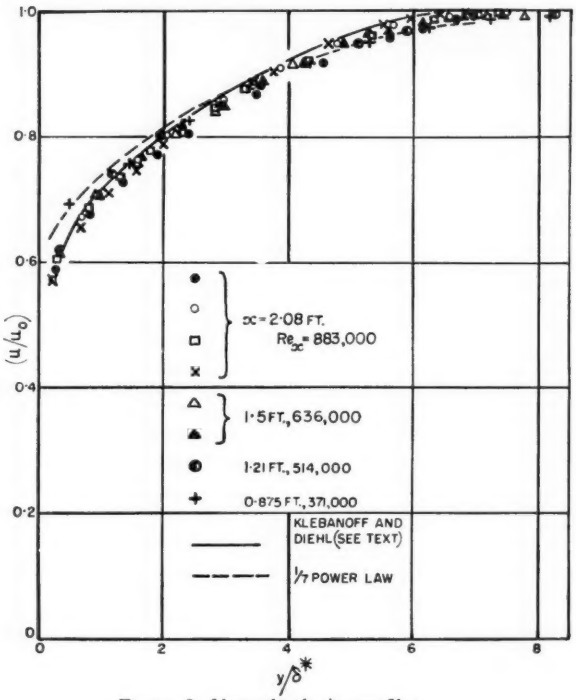


FIGURE 2. Normal velocity profiles.

plate surface; the last term of equation (2) then vanishes. The quantity and velocity of the added fluid was measured using a $\frac{1}{4}$ radius Pitot tube flow meter[5]. To promote a turbulent boundary layer, a trip wire transition device was glued to the plate surface near the rounded leading edge. The diameter of this wire was calculated from Reference 6. Provision was made for velocity traverses at various stations downstream of the point of air addition.

NOTATION

$\delta$ boundary layer thickness (ft.)

$\delta^* = \int_0^{\delta} u\,(u_0 - u)\,dy$ boundary layer displacement thickness (ft.)

$F$ total frictional drag (lb.)

$u$ velocity (ft./sec.)

$\rho$ mass density (slugs/ft.$^3$)

$w$ mass of fluid added per unit time per unit width of plate surface (slugs per sec./ft.)

$y$ wall distance (ft.)

$T$ shear stress (lb./ft.$^2$)

$x$ distance (ft.)

$\Delta x_1$ length over which fluid is added (ft.)

$\theta$ angle between plate surface and axis of fluid addition

$\mu$ absolute viscosity (lb. sec./ft.$^2$)

$c_D = F/\frac{1}{2}\rho u_0^2$ drag coefficient

$\left(\frac{u_1}{u_0}\right), \left(\frac{u_1}{u_0}\right)\left(\frac{\Delta x_1}{x - x_1}\right)$ added fluid quantity parameters

$\frac{u_0 x \rho}{\mu}$ Reynolds number

*Suffixes*

1 fluid addition

0 mainstream

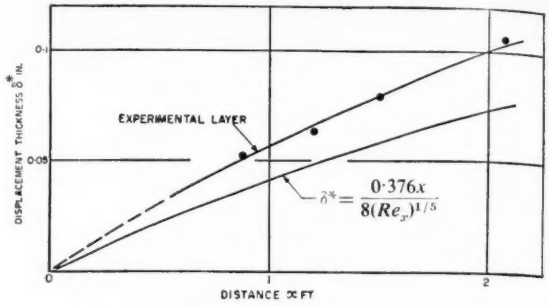


FIGURE 3. Plate boundary layer.

EXPERIMENTAL RESULTS AND DISCUSSION

To establish that a fully developed turbulent boundary layer existed under normal flow conditions (*i.e.* no secondary air), velocity traverses were made at 4 stations, 0·875 ft., 1·21 ft., 1·5 ft. and 2·08 ft. along the plate. The results of typical tests are shown in Fig. 2. The displacement thicknesses of the boundary layer have been computed by numerical integration. On the whole, the profiles are in good agreement with one another. The slight differences between the profiles for the 4 sections are no greater than those for the same section. The mean turbulent boundary layer profile of Klebanoff and Diehl[4] for free transition at a wind speed of 108 ft./sec. and the seventh power law have been plotted for comparison. The present results indicate that the transition device employed was entirely satisfactory in producing a fully developed turbulent boundary layer early in the flow.

The experimental layer, produced with the transition device, thickens more rapidly than the normal layer (based on the 1/7th power law and the assumption that the layer is turbulent right from the leading edge) which can be calculated from

$$\delta^* = \frac{\delta}{8} = \frac{0{\cdot}376x}{8\,(Re_x)^{1/5}} \quad . \quad . \quad \text{(see Ref. 1)}$$

Fig. 3 shows this artificial thickening.

The experimental data for the flow with air addition is shown in Figs. 4, 5 and 6 and in Figs. 7, 8 and 9 in non-dimensional form which is the better basis for comparison. The following observations have been made:—

(*i*) There is a further thickening of the boundary layer.

(*ii*) For a given value of $(u_1/u_0)$ the profile becomes fuller in the $x$-direction.

(*iii*) At the farthest sections, the profile becomes fuller as $(u_1/u_0)$ increases and eventually becomes more square than the normal one.

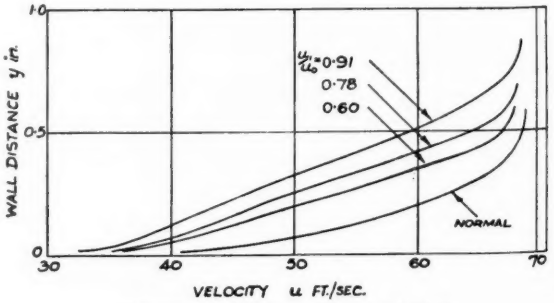


FIGURE 4. Velocity profiles. $x = 1{\cdot}21$ ft.

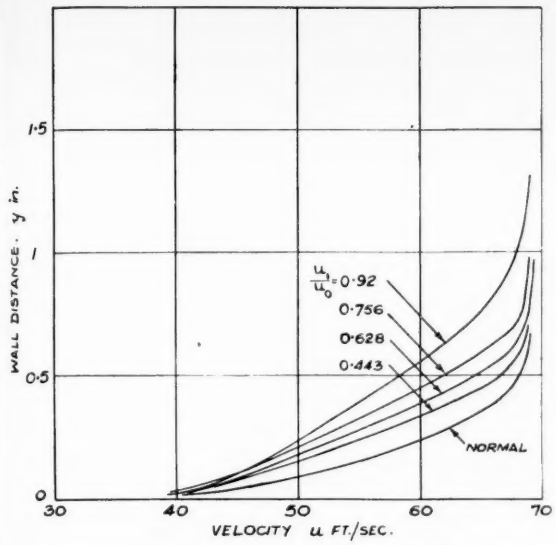


FIGURE 5 (*left*). Velocity profiles. $x = 1{\cdot}5$ ft.

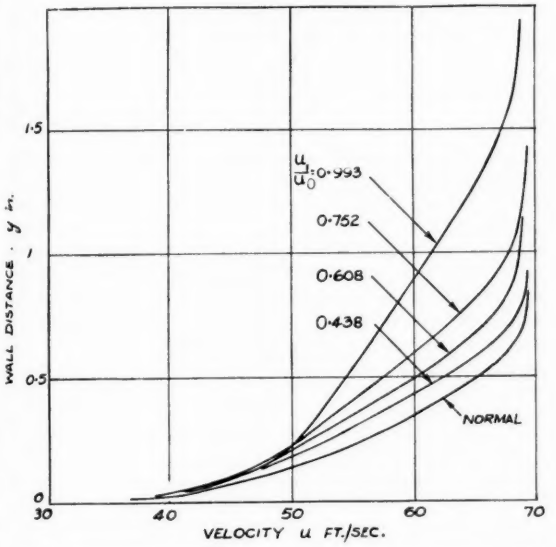


FIGURE 6 (*left, centre*). Velocity profiles. $x = 2{\cdot}08$ ft.

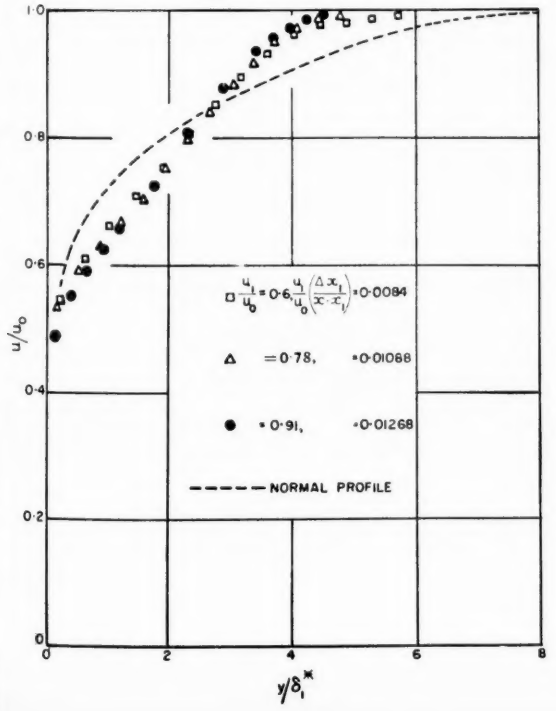


FIGURE 7 (*left*). Velocity profiles with air addition. $x = 1{\cdot}21$ ft.

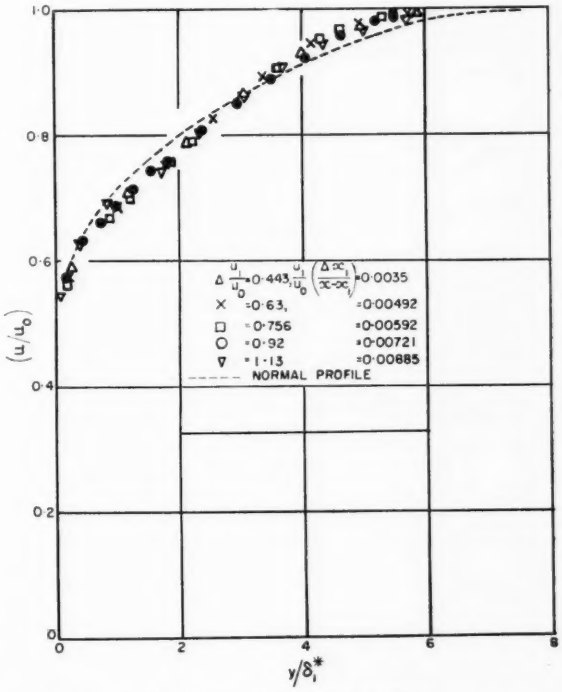


FIGURE 8. Velocity profiles with air addition. $x = 1{\cdot}5$ ft.

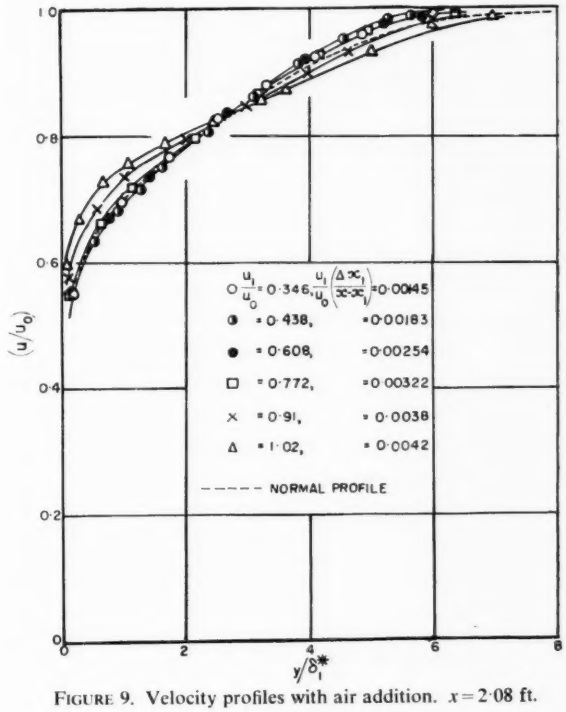


FIGURE 9. Velocity profiles with air addition. $x = 2{\cdot}08$ ft.

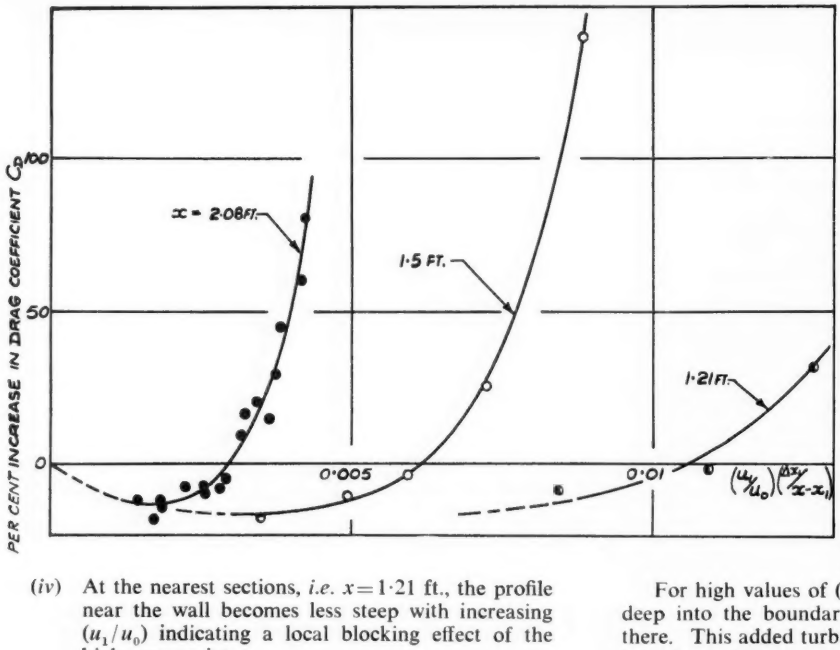


FIGURE 10. Plate drag.

(*iv*) At the nearest sections, *i.e.* $x=1{\cdot}21$ ft., the profile near the wall becomes less steep with increasing $(u_1/u_0)$ indicating a local blocking effect of the high energy jet.

Evidently, the velocity distribution is affected by the combined effects of the penetration of the jet and the turbulent mixing of the main and secondary airs as the flow proceeds. (In an experimental study on the friction and flow characteristics on a rough porous surface, Eckert, Diaguila and Donoughe[2] point out the possibility of added turbulence due to the jet action of the surface pores which would tend to increase the friction.) Immediately downstream of the orifice, in a region close to the surface, there will be a reduction in velocity which becomes more pronounced the greater $(u_1/u_0)$. Proceeding in the flow direction the added turbulence begins to spread through the layer resulting in a squarer profile. At sufficient distance downstream and at sufficiently high injection rates the profile eventually becomes more full than the normal one.

These velocity observations assist in the explanation of the effects on the local and total frictional drags; since if

$$T = \mu \frac{du}{dy}$$

$$\text{then} \quad T \propto \left[\frac{d(u/u_0)}{d(y/\delta^*)}\right]$$

which is a more suitable form for use with the non-dimensional profiles.
At the wall,

$$T_w \propto \left[\frac{d(u/u_0)}{d(y/\delta^*)}\right]_{y=0}$$

## PLATE FRICTION

The frictional drags for the whole surface have been calculated using equations (1) and (2). Three plate lengths have been tested and the results are shown in Fig. 10. Because of the large number of possible variables the results are qualitative only. The parameter

$$\left(\frac{u_1}{u_0}\right)\left(\frac{\Delta x_1}{x - x_1}\right),$$

indicating the amount of air addition, is analogous to the blowing quantity coefficient used in the literature on boundary layer control (*see*, for example, Goldstein[3]).

It was noted that in all cases the total friction first decreased and then increased rapidly. At the lower end of the scale, the secondary air is confined to flow in a narrow region adjacent to the plate surface, and as its initial velocity in the flow direction is zero, the velocity near the boundary surface is reduced with a corresponding decrease in wall shear stress. This was ascertained by direct velocity measurements with the Pitot tube stagnation point in contact with the plate surface. At all three sections, there is a decrease in velocity for small amounts of fluid addition.

For high values of $(u_1/u_0)$, the secondary air penetrates deep into the boundary layer causing greater turbulence there. This added turbulence spreads towards the wall and into the mainstream resulting in a rapidly increasing local wall shear stress in the $x$-direction. In the wall position at $x=1{\cdot}21$ ft., the Pitot tube recorded a velocity smaller than that for small air additions, but at the farthest section the velocity was greater, these observations being in agreement with the general velocity measurements. Evidently then, for high rates of air addition there is a decrease in local friction just downstream of the orifice but an overwhelming increase at the farthest sections, the net effect being an increase in the total friction above that in normal flow. There is a secondary air flow rate for a given plate length for which there is no effect on the total drag.

It appears that local fluid addition is not a satisfactory means of artificially thickening a turbulent boundary layer.

## ACKNOWLEDGMENT

The author wishes to thank Professor W. J. Kearton for permission to use the Dynamics Laboratory and the Engineering Workshop in the Department of Mechanical Engineering of the University of Liverpool.

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# *Safety and Large Aircraft*

WALTER TYE, O.B.E., B.Sc., F.R.Ae.S.

*(Chief Technical Officer, Air Registration Board)*

MR. CRAWSHAW (p. 831 December 1957 Journal) believes that it was a grave error of the aircraft industry to accept a factor of 2 for pressure cabins, and that this factor should "have been immediately increased officially after the Comet disaster." He feels that the requirement for a satisfactory fatigue life is somewhat futile for, as he says, "Who knows whether any given pressurised fuselage has a satisfactory fatigue life?"

As regards achieving satisfactory fatigue life by an arbitrary increase of static strength factors, I would suggest that one of the few clear pointers from recent experience is that such a process is unlikely to be successful. To pose another question, "Who knows what static factor to apply to achieve satisfactory fatigue lives?" In a recent accident it seemed likely that fatigue failure had occurred in a part in which there was a static factor of about 6 on the nominal repeated loads. Would a factor greater than 6 be acceptable? I doubt it.

It is, of course, true that pre-war aeroplanes escaped catastrophic fatigue failure, and this may have been due in part to the higher static factors then applied. But there were other features of these older aeroplanes. For instance, the materials used were a good deal less sensitive than current high tensile light alloys and steels. Also the aeroplanes concerned, being smaller in size, were to a greater extent the product of small experienced design teams, with well developed instincts for what constituted good design. It is doubtful whether, with the large present-day drawing offices, the touch of the experienced hand is so fully present; and a single piece of inept detail design is sufficient to reduce fatigue life ten-fold. Thus I can only accept the pre-war reliance on static factors to provide good fatigue properties, if Mr. Crawshaw can accept the other conditions which then applied.

On the subject of establishing the value of a safe life of a component, if Mr. Crawshaw's despondent question represents the attitude of designers, the outlook is indeed a dim one. Unless the structure is fundamentally fail-safe in character, safe lives must be established. Remember that in six years British civil aeroplanes have suffered several catastrophic fatigue accidents, as well as three very near misses. I do, however, sympathise with the view that in order to be safe, penalising life factors have to be applied, and expensive test programmes undertaken.

Thus we come to Mr. Crawshaw's support for the fail-safe approach, with which I generally agree. It is an odd thing that in no other field than the structural one do we rely on single elements to provide safety. The structure is nearly good enough to do so, but not quite. Acceptance of the fail-safe concept, and the development of design and test techniques to establish that a structure is generally fail-safe, will in the end remove the difficulties which Mr. Crawshaw so rightly emphasises.

*Received 31st December 1957.*

WILLIAM BROOKES, D.C.Ae., Grad.R.Ae.S., G.I.Mech.E.

*(Initial Projects Office, A. V. Roe and Co. Ltd., Chadderton)*

A. H. CRAWSHAW shows a certain lack of appreciation of the applicability of probability theory to the design of aircraft.

In the first place the statement that "if an aircraft carries hundreds of people its safety is hundreds of times more important than that of an aircraft which carries only a few" is misleading, since it is false for a given total carrying capacity of a fleet of aircraft—the only reasonable basis for comparison.

The suggestion that the requirement for a satisfactory fatigue life for components should be supplemented by an increase in the safety factor for static strength is ludicrous especially in view of the reference which was made to boiler practice. The only justifiable criticism that can be levelled at the existing requirements is that no lead is given as to how it can be endorsed that they are actually met.

It is now common practice to multiplicate components where practicable, in order to increase safety in case of failure. Nevertheless, due to economic and weight considerations, there are limits to which the principle can be extended. Even where duplication is feasible there is still a finite possibility of failure of two components in one flight or within the same period between inspections. Thus probability theory is still applicable in estimating the likelihood of some catastrophic failure taking place in the aircraft as a whole.

In summary it may be stated that no aircraft can be made 100 per cent safe. It can only be designed in such a way that the probability of catastrophic failure is estimated to be less than a chosen value. This value would be governed directly or indirectly by balancing the importance or value of a representative cargo against the cost of aircraft and operation.

*Received 13th January 1958.*

DONALD HOWARD, A.F.R.Ae.S.

ALTHOUGH I am nowadays something of an outsider, I am inclined to agree with the preliminary remarks of Mr. A. H. Crawshaw's note in the December Journal. I know from my experience within the Aircraft Industry that for the majority of the members of the Society, most of the articles in the Journal are well nigh unintelligible. As Mr. Crawshaw says, to be able to understand most of the articles, one has to be a mathematical genius, and the percentage of the members of the Society who might be described as "mathematical geniuses" is, I imagine, very small. It wouldn't surprise me if there are still a large number of members of the Society with an intense interest in aircraft and aviation. Such members would be grateful, I feel, for a few more articles of historical and general interest in addition to the highly technical papers which I know have an extremely important and necessary place in the Journal.

*Received 6th January 1958.*

# Graduates' and Students' Section

## Techniques and Procedures in Supersonic Flight Testing

MR. R. P. BEAMONT, Manager of Flight Operations and Chief Test Pilot to the English Electric Co. Ltd., spoke to the Section on 27th November 1957.

He explained that until very recently the problems of supersonic flight testing had not been faced in this country, and that they had had to develop new techniques to meet them.

*R. P. Beamont*

The first stages of the test programme were the same as for any aeroplane, and flight limits were set to avoid entry into a critical flutter region. This could still allow high Mach numbers if there were no Mach/flutter limit.

It is in this initial stage that the use of "chase" aircraft as is done in the U.S.A. would be most valuable, if one of comparable performance were available, since minor snags, such as undercarriage door retraction, may develop where a second pair of eyes, able to see the outside of the aeroplane, may reduce hazards and save valuable flight time.

Once initial handling is complete and the services are adequately reliable the flutter limits are raised to approach the design limits. To excite vibrations without flying through turbulent air or "stick jerking," a new method developed by the R.A.E. was used; known as "developed bonking," it consists of firing a small explosive charge at right angles to the surface. The degree of bonking being known, the recordings of damping can be compared with the calculated values.

Although enabling the rate of flutter testing to be speeded up it did not prevent running into fin-rudder flutter precisely between one stage and the next. This was cured by altering the fin stiffness, and, as an added safeguard, fitting a viscous rudder damper.

The next major problem experienced was due to the vast distances covered during high speed runs, and the difficulty of keeping out of airways and ensuring that flight is only supersonic over the sea or when facing away from the coast. Mr. Beamont agreed with a questioner that a very good case could be made for high speed flight testing in some other country away from the restrictions of weather and geography.

These restrictions, and the lack of precise navigational information still caused the greatest difficulty when planning test sorties, as the endurance on re-heat was not great enough to allow anything to be left to chance. As an example of navigating difficulties, Mr. Beamont quoted an average time of two minutes between asking for, and getting a VHF fix, during which time the aircraft might travel 50 miles!

Once flutter problems are cleared the stage is set for measuring stability, control and performance up to design Mach number. Attention had been focused by experiences in the U.S.A. on the characteristic reduction in the direction stability derivative $N_v$, which can occur from about $1{\cdot}4M$, and speed was increased by increments of $0{\cdot}05M$ from $1{\cdot}2M$ and a rudder displacement force of 150 lb. was applied and released. $N_v$ remained positive and all was well. The aircraft could be trimmed hands and feet off at these tremendous speeds, and even flown at large angles of sideslip and left to sort itself out.

Care had to be taken on other counts, since the region where intake buzz might occur was being approached. Change of trim with transition from subsonic to supersonic flight was scarcely noticeable unless under $g$, when a total increment of $\frac{1}{2}g$ might be felt.

Summing up Mr. Beamont said his main impressions were:—

1. Satisfaction in an unusually well harmonised pilot's aeroplane at all speeds.
2. Frustration in the continual difficulties experienced in navigating without adequate navigational facilities.
3. Embarrassment over inability to carry out tests at the speeds required without causing shock wave incidents.

### Visits

Visits have been arranged to the Lotus Engineering Company, Hornsey, N.8, on Friday 28th February at 7.30 p.m. and to B.O.A.C. at London Airport on a date to be announced in April. The last Lotus visit was very popular and to avoid disappointment early application is advised to the Hon. Visits Secretary, Mr. N. K. Benson, 14 Wakering Road, Barking, Essex.

### Annual General Meeting

The Section's Annual General Meeting will be held at 4 Hamilton Place on Tuesday 25th March 1958 at 7.30 p.m. Light refreshments will be served from 7.00 to 7.30 p.m. The meeting will be followed by a film show on a variety of subjects of aeronautical interest.

Nominations for the Section representatives on the Committee for the year 1958-59 are invited, and should be sent to the Hon. Secretary:—

Mr. P. D. Stewart
10 Abbey Gardens, Chertsey, Surrey.

All nominations must be both proposed and seconded, with the candidate's agreement, and all concerned must be Graduates or Students. The number of vacancies to be filled is nine and, if the nominations exceed this number, a ballot will be held during the meeting.

The successful candidates' names will be subject to confirmation by Council.

Members are reminded that it is desirable for the Committee to be representative of as many firms or colleges as possible.

# THE LIBRARY

## *Reviews*

FLIGHT TODAY. *J. L. Naylor and E. Ower. 4th Edition. Oxford University Press, 1957. 184 pp. Illustrated. 12s. net.*

The fact that this book, first published in 1936, has now reached its fourth edition and that there have also been three reprints of earlier editions, shows that it has met a demand for a non-technical volume giving the ordinary reader some account of the progress made in the conquest of the air. This latest edition has been considerably extended and brought up to date as far as security regulations allow. It deals briefly with the design and operation of aircraft and their power plants and with such related subjects as learning to fly and the effects of the weather on flight.

It is well known to be one of the most difficult tasks to write a book on a scientific subject in non-technical language. If the author is well versed in his science he is quite likely not to be able to achieve a first class literary style, while if he is a literary genius he is not likely to know his science well enough to avoid serious errors. The combination of both desirable qualifications is a rarity indeed and few writers have achieved it. The present volume seems to fall into the first category; the facts, as stated in it, and they are rather too many than too few, are reasonably free from error, but the literary style can hardly be described as very good. One does not, for instance, easily pardon such tautologies as "heavier weight" or "slower speed." However, one must not judge a very difficult task too harshly, and the book can undoubtedly give the layman a very good picture of what has been done and how it has been accomplished. Perhaps less space might have been devoted to balloons and parachutes, and used instead for a simple explanation of the way in which aircraft instruments work; the reader does not even learn how air speed is measured.

The almost incredible rapidity with which aeronautics has developed, especially in the last decade, is very well brought out in a number of ways in the course of the book, and there seems little doubt that the authors will be able to contemplate yet another edition before many years have passed!—ERNEST F. RELF.

CLOUD STUDY—A PICTORIAL GUIDE. *F. H. Ludlam and R. S. Scorer. John Murray, London, 1957. 80 pp. Illustrated. 12s. 6d.*

This beautifully illustrated book is sponsored by the Royal Meteorological Society and includes many photographs from the Society's collections. Perhaps this explains the remarkably low price. The authors have added photographs of their own and have chosen them, they explain, not as examples of the standard classifications, but to show the dynamics of cloud formation and dissipation.

If the reader is tempted, when studying a beautiful cloud photograph, to think of a cloud as static he is corrected by the description appended by the authors. They describe six processes by which clouds are formed, seven by which their character may be modified and then explain how evaporation eventually destroys the cloud. The seventy-five photographs are each followed by a detailed description of the processes seen at work in the picture.

The authors have an interest in flying which is exceptional even among meteorologists and many of the photographs reflect this. The wave clouds of special interest to glider pilots are well illustrated and the condensation trails produced by aeroplanes are discussed. The complex physical processes in cloud formation are carefully described without resort to mathematics. The complexity is well illustrated by the remark that, "In the atmosphere it is found that out of every million cloud droplets only one is likely to be frozen at a temperature of $-10°C$ and only a few hundred at $-30°C$." A belief that the freezing point of water is $0°C$ is clearly inadequate.

Messrs. Ludlam and Scorer and their colleagues in the Department of Meteorology at Imperial College have played a leading part in cloud research. Those who fly among clouds and those who enjoy watching them from the ground will value this excellent little book.—A. H. YATES.

THERMAL STRESSES. *B. E. Gatewood. McGraw-Hill, New York, 1957. 232 pp. Diagrams. 56s. 6d.*

During the past decade the structural problems associated with the kinetic heating of aircraft have exercised the ingenuity of the aeronautical engineer and a multitude of research reports have been published. The diversity of this information is such that a newcomer to the subject is at a loss to find a starting point. This book adequately deals with this difficulty. Here in one volume is an orderly statement of the structural problems associated with temperature effects, beginning with elementary schoolboy physics and continuing through to recent advanced work. Nor is the subject matter of the book confined to that implied by its title, since it concerns itself with the aspect of material properties at high temperature, as well as the stresses resulting from transient heating conditions.

This is no purely academic treatise, as might be implied by the term "review," but is obviously the work of one who has personally encountered the practical difficulties of the subject. Blatant statements of dubious merit are carefully avoided, and caution in the use of theories is recommended where this is deemed to be necessary. Many analytical procedures are outlined and the understanding of them greatly assisted by extensive worked examples—together with numerous unsolved problems for the enthusiastic! There is also an extensive and comprehensive bibliography at the end of each chapter.

Rather disappointingly the section concerned with design recommendations is less than one page in length, and this contains the only specific mention of the problems associated with the use of thermal insulation. Even this one reference is somewhat vague.

By and large the book concerns itself with methods rather than answers. As such it should be within the grasp of anyone concerned with thermal stresses, whether he be a student just meeting the subject for the first time, or an experienced research worker.—D. HOWE.

BORON, CALCIUM, COLUMBIUM AND ZIRCONIUM IN IRON AND STEEL. *R. A. Grange and others. John Wiley and Sons, New York, 1957. 533 pp. Illustrated. 112s.*

About 3,000 years ago the Iron Age began. Initially and for long subsequently, although the metal is over seven hundred times more prevalent in the Earth's crust than its ancient and far earlier established rival, copper, judged on an output basis, it occupied an inferior position. Now, however, since iron production is over seventy times greater than that of the older metal, it deserves the dictum, of Kipling's Baron when speaking of the metals, "Iron—cold iron—is master of them all."

During the metallurgical age in which we live, Man,

generally, without knowing it, has invariably used, not pure iron, which has a hardness of only about 50 V.P.N., but an iron alloy and, during the past hundred years he has more and more turned to that class which is termed steel. This, a work of reference, with accuracy rather than with elegance, defines as "that form of iron produced in a fluid condition, and hence practically free from slag (difference from wrought iron) which contains less than about 2·2 per cent of carbon—as a rule less than 1·5 per cent (difference from cast iron)."

Today, such steels are very extensively used, though scarcely at all in aircraft. In this field, as in many others, there is a demand for much higher mechanical properties than they possess both at normal temperatures and also at elevated ones. As a consequence, the meeting of such requirements has necessitated the use of alloy steels. These come within the terms of the quoted definition, but they contain one or more additional elements.

Chronologically, Hadfield's 12-14 per cent manganese alloy of the early eighteen eighties should first be mentioned. Then shortly followed steels, into which had been introduced what is probably the most valuable of all alloying elements, nickel. At the beginning of this century came tungsten, chromium, vanadium and molybdenum, each of which conferred valuable properties. Then, immediately before the First World War, Brearley, by the addition of 12-14 per cent of chromium, produced his revolutionary rustless steels and finally, largely since the Second World War, have come into being the steels which form the subject of the book now under consideration.

This work by Grange, Shortsleeve, Hilty, Binder, Motock and Offenhauer—a combination of names somehow suggesting a selection of Uncle Tom Cobbleigh's riding companions—deals with steels containing boron, calcium, columbium and zirconium. It is composed of a series of very good papers, well reproduced and, in most cases, supplied with excellent bibliographies and equally good indices. The reviewer having said what he has, with regret but he hopes with wisdom, does not recommend this book, costing rather over five pounds sterling a copy, to the Aircraft Industry. The airframe engineer and metallurgist would derive much sound information from it but, for them, it would have no practical applications. Their gas turbine counterparts would largely be in a like case. They could, however, read with interest, but not always in complete tranquility of mind, some sixty pages on ferrous alloys mainly containing niobium (the internationally agreed name for what in this book is called columbium) intended for gas-turbine applications, many of which are, however, met and perhaps even better met by the use of non-ferrous alloys.

This erudite compilation can and should claim immediate entry into a metallurgical library, but it would be mere politeness to let it into an aeronautical one. Here, one suspects, it would be as neglected as Cinderella before the arrival of the Fairy Prince.—P. T. TEED.

THE GEOGRAPHY OF AIR TRANSPORT. *Kenneth R. Sealy. Hutchinson University Library, London, 1957. 207 pp. Illustrated. 10s. 6d.*

There is a growing bibliography on the subject of The Geography of Air Transport. Two early volumes, *The Geography of World Air Transport* by J. Parker Van Zandt 1944, and *Air Transport and Geography* by W. G. V. Balchin 1947, and the recent volume by M. Eugéne Pepin, Director of the Institute of International Air Law at McGill University, *Geographie de la Circulation Aérienne*, are already on the shelves of the Society's library. This new volume is therefore in good company. It is one of the Hutchinson University Library series and the author, who is lecturer in Geography at the London School of Economics, covers a field extending from the technical and economic background to the effects of physical geography on the air transport patterns in Europe, the U.S.A. and Canada.

The influence of the physical world on the air transport business is perhaps never far from our thoughts, since it is this that helps to determine our designs and their economy at every stage and defines our route system. Mr. Sealy is too wise a geographer to confine his study to map projections and statistics of populations. He writes effectively on such topics as the influence of ambient air temperature on the turbo-jet airliner and provides effective short examples of many of the operational problems in air transport which will be of value to students of air transport economics and are not readily available in other recent books on the subject.

It would not be possible in a book of 200 or so pages to cover the broad field chosen by the author in great detail. This reviewer, however, found it difficult to pick up serious errors in omission. The author's great interest in the Canadian field is well used as the background for examples on air transport in under-developed areas. This may slightly limit the scope of the work, but a fuller treatment would have required a much larger volume.

There is a great need for more books of this kind on air transport subjects. It is most moderately priced, is well illustrated and provides up-to-date examples. The excellent references and bibliography will be valued not only by students of the subject.—A. H. STRATFORD.

THE ROCKET PIONEERS. *Beryl Williams and Samuel Epstein. Lutterworth Press, London, 1957. 247 pp. Illustrated. 15s.*

This most interesting—and today most topical—book is undoubtedly the result of considerable research on the part of the authors. While the broad lines of many of the biographies contained in "The Rocket Pioneers" are available in other places, this is the first time that they have been collected under one cover, and also related in a sympathetic and understanding manner. Perhaps in this year of progress, when so much of the working thought of these men has been crystallised into the reality of at least the first step toward space travel, it is too easy to smile at the lack of perception of those who opposed not only their work, but also the basic theories of the rocket vehicle and its power plant.

It is a most sobering thought that in the first half of this century of scientific and technological improvement many of the achievements were initiated in primitive conditions usually outside the facilities of allied industries. It is only necessary to think of the early days of flying, or the gas turbine, to realise how easy it is for established technology to fail to recognise valid new concepts.

This, then, really is a book about pioneers, men who, for so many reasons, moved against the tide of established thought. Theoretical men like Ziolkovsky and Oberth, the experimental groups within both the German and American Rocket societies of the early 1930's—enthusiasts, lacking often in funds and scientific discipline but never in ideas and verve. Then Goddard the physicist, who sought the means to reach high altitude and of that necessity became chemist, engineer, fitter and designer of his own engines, vehicles, test facilities, and instrumentation. Also Congreve who as an artillery man, son of an artillery man, displaced the smooth bored cannon with solid propellant rocket artillery, one hundred and fifty years ago. Here, too, is the pre-war British Interplanetary Society, prevented by the Explosives Act of 1875 from following the practical

trends of the German and American Societies, laying down instead a foundation of theoretical enquiry that continues today.

Together with the text there are some 30 illustrations making together a well balanced work.—D. S. CARTON.

PROCEEDINGS OF THE SOCIETY FOR EXPERIMENTAL STRESS ANALYSIS. *Vol. XV. No. 1. Mahlmann and Murray (Editors). Cambridge, Mass., 1957. 200 pp. Illustrated.*

The eighteen papers in this number of the Proceedings cover a wider range of subjects for investigation and stress analysis methods than usual. The subjects include fatigue, radiant heating, residual stresses, recording the motion of an oil-well drill at a depth of 2,000 ft. and the stresses in threaded connections. New work on established methods of stress analysis is represented by papers on strain gauges and on brittle coatings (one on the dynamic properties of Stresscoat), while there is also a paper on the photoelastic method of checking the residual stresses in the glass bulbs of thermionic valves. Among the less well-known techniques for the measurement of surface strains is the photoscreen method, which uses mechanical optical interference, and its advantages for the presentation of a general picture of the strains is emphasised in an Australian paper. The use of multiple-exposure still photography for recording large transient displacements is recommended in another paper. It is shown elsewhere that the ratio of two transducer outputs can be recorded directly through a modified Wheatstone bridge, provided that the transducers are of the variable resistance type. Another paper describes a way of solving Laplace's equation numerically for a structural member with an irregular boundary; star networks are used in some parts of the field and square networks in others.

The subject chosen by Lazan for the 1956 William M. Murray lecture was that of fatigue under resonant vibrations, which has many applications in the aircraft field. In the absence of special damping devices the amplitude of vibration is limited in two ways, by internal damping in the material and by external damping such as occurs at the surfaces of a joint. Both kinds of damping may be important, but it can be said that internal damping generally plays a greater part when the stresses are high whereas the effect of external damping is larger at intermediate and low stresses. Methods of test designed to represent the aerodynamic heating effect associated with supersonic flight are of particular interest. In a description of the radiant heating equipment at the Massachusetts Institute of Technology, the simulation of temperature and thermal stress responses with scale models is discussed. Test results are included on a light alloy box beam heated on one side and on both sides and also on a model titanium wing heated on both surfaces. A useful contribution from the General Motors Technical Center, Detroit, is on a comparison of the X-ray diffraction method with dissection methods for the measurement of residual stresses in hardened steel. It is concluded that the accuracies of the two methods are comparable and that X-rays are particularly useful when the main interest is in surface stresses. An interesting suggestion from Japan is that the stress in a plate can be estimated by applying a lateral impulsive force to it and then recording the time-displacement curve some distance away from the point of application of the force.—A. F. C. BROWN.

## *Additions to the Library*

**Aviation Electronics Technician 1 and C.** Bureau of Naval Personnel. U.S.G.P.O. 1957. 27s. One of a series of Service instruction books.

**Conference Terminology.** Edited by Jean Herbert. Elsevier Publishing Company, Amsterdam. D. Van Nostrand Company Inc., New York. Cleaver-Hume Press Ltd., London. 1957. 147 pp. 12s. 6d. A tabulated glossary of 750 words and phrases; subtitled "A Manual for Conference-Members and Interpreters." In English, French, Spanish, Russian, Italian, German.

**Dynamic Instability.** J. Rocard. Crosby Lockwood. 1957. 218 pp. 45s. A translation from the French work by a Professor in the University of Paris and Director of the Physics Laboratory of the Ecole Normale Superieure. Rocard has studied stability problems as applied to road, rail and air transport and to suspension bridges. He analyses the famous Tacoma bridge disaster in the last context.

**Electronic Engineer's Reference Book.** L. E. C. Hughes (*General Editor*). Heywood. 1957. 1,303 pp. 84s. Perhaps the one branch of engineering which has not hitherto had its handbook is electronics. The rapidity of the branch's growth is shown by this volume of 1,303 pages, with its 98 sections and 989 subsections. Fifty authors have combined to produce what will probably be the standard reference book on the subject—a subject which has applications in all industry.

**History of German Guided Missiles Development.** *Agardograph 20.* I. Benecke and A. W. Quick (*Editors*). Verlag E. Appelhaus. 1957. 55s. *To be reviewed.*

**Insect Flight.** J. W. S. Pringle. C.U.P. 1957. 132 pp. 15s. Number 9 of the Cambridge Monographs in Experimental Biology it covers the mechanics of wing motion, the structure and physiology of flight muscle, aerodynamics and nervous and sensory mechanisms. The section on Aerodynamics is essential reading for anyone looking into the aerodynamics of insect flight.

**Making of a Moon, The.** A. C. Clarke. Muller. 1957. 21s. *To be reviewed.*

**IX Congrès International de Mechanique Appliquée-Actes.** Tome II. University of Brussels. 1957. The Second Volume of the proceedings of the 9th Congress on Applied Mechanics. These are being reviewed.

**Radiosonde Data Processing Computor, A.** (*Cook Technical Review, Vol. 4, No. 2.*) R. Payne. Cook Research Laboratories. 1957. 1 dollar. A special number of a house journal describing the application of the computer in processing radiosonde data.

**Rockets, Missiles and Space Travel.** W. Ley. (Second Edition.) Chapman and Hall, London. 1957. 520 pp. Illustrated. 50s. A revised and enlarged edition of a well-known book which was first published in this country in 1951.

**Scorpion Canberra Altitude Record, The.** D. Napier and Son. 1957. A pamphlet on the record of 28th August 1957.

**Simulator for Use in Development of Jet Engine Controls.** *N.B.S. Circular 584.* E. S. Sherrard. U.S.G.P.O. 1957. 1s. 10d. Describes a simulator intended for use in determination of the stability and performance of the engine control system.

**Soaring Pilot.** Welch, Ann and Lorne, *et al.* London. John Murray. 1955. 18s.

**Sources of Invention, The.** John Jewkes, *et al.* Macmillan. 1958. 31s. 6d. *To be reviewed.*

**Three Steps to Victory.** Sir Robert Watson-Watt. Odhams Press. 1958. 30s. *To be reviewed.*

**Water Waves.** J. J. Stoker. Interscience Publishers, New York. 1957. 567 pp. Illustrated. 12 dollars. Volume IV in the Interscience Series "Pure and Applied Mathematics." The mathematical theory of wave motion in liquids, with its applications—floods in rivers—tides—waves on a beach—motion of a ship—and many others.

# Reports

## AERODYNAMICS

*See also* MATHEMATICS

BOUNDARY LAYER—*see also* INTERNAL FLOW
THERMO-AERODYNAMICS

*Effects of extreme surface cooling on boundary-layer transition. J. R. Jack et al. N.A.C.A. T.N.* 4094. *October* 1957.

Boundary-layer transition studies were made on various bodies of revolution at a Mach number of 3·12 and for Reynolds numbers per ft. as large as $15 \times 10^6$.—(1.1.2.4 × 1.9.1).

COMPRESSIBLE FLOW—*see also* THERMO-AERODYNAMICS
INTERNAL FLOW
WINGS AND AEROFOILS

*Molecular approach to problems of high-altitude, high-speed flight. G. N. Patterson. U.T.I.A. Review* 10. *May* 1957.

To obtain a better understanding of the aerodynamics of high performance aircraft, the effects of rarefaction and high temperature on the flow of gases must be investigated. A review of these effects is given and the increasingly important role played by the molecular structure of a gas is indicated.—(1.2.3).

*Study of pressure distributions on simple sharp-nosed models at Mach numbers from* 16 *to* 18 *in helium flow. W. D. Erickson. N.A.C.A. T.N.* 4113. *October* 1957.

Experimental pressure distributions have been obtained on some simple sharp-nosed aerodynamic shapes in helium flow at Mach numbers from 16 to 18. Temperature recovery factors are also presented. The results are compared with theory which accounts for the shock-boundary-layer interaction. Tunnel-calibration results are also shown.—(1.2.3).

*A rapid method for predicting attached-shock shape. E. S. Love and R. H. Long. N.A.C.A. T.N.* 4167. *October* 1957.

A method is presented for the rapid prediction of the shape of attached shocks emanating from smoothly contoured axisymmetric and two-dimensional nose shapes. From a practical viewpoint the accuracy of the method is comparable to that of the method of characteristics.—(1.2.3.2).

CONTROL SURFACES

*A geared flywheel balance arrangement for the prevention of control surface flutter. W. G. Molyneux. C.P.*365. 1957.

Some geared mass balance systems are described that are effective in eliminating inertial couplings between modes of the main surface and the control surface. These arrangements are shown to have an advantage over the conventional arrangement for preventing flutter involving main surface torsion and control rotation, particularly where there is a near frequency coincidence between the modes.—(1.3 × 2).

*A flight technique for the measurement of thrust boundaries and of drag due to lift. H. D. Rylands. AGARD Report* 123. *May* 1957.—(1.3.3).

FLUID DYNAMICS

*The calculation of the paths of vortices from a system of vortex generators, and a comparison with experiment. J. P. Jones. C.P. No.* 361. 1957.

A method is described for the calculation of the paths of the trailing vortices from a system of counter-rotating vortex generators, and a comparison is made with some experimental results obtained in a wind tunnel. The method can be adapted to other configurations of generators arranged to produce rows of vortices close to a plane surface.—(1.4).

*An experimental investigation of the oscillating pressures on a circular cylinder in a fluid stream. D. M. McGregor. U.T.I.A. T.N.* 14. *June* 1957.

Results of experimental investigations of the fluctuating pressures at the surface of a circular cylinder are presented. The cylinder was mounted in the U.T.I.A. subsonic wind tunnel at right angles to the flow and contained within it the microphone used as the pressure transducer. An attempt to develop a simple mathematical model of the flow by considering an alternating vortex standing at the rear of the cylinder met with moderate success. Estimations were made of the fluctuating lift and drag coefficients of the cylinder.—(1.4).

INTERNAL FLOW—*see also* SCIENCE—GENERAL
THERMODYNAMICS

*The flow through short straight pipes in a compressible viscous stream. J. Seddon. C.P.*355. 1957.

The principal characteristics of compressible flow in simple hollow pipes at zero yaw and a theory for calculating the effect of the boundary-layer are given. This is checked against the results of tests with a series of pipes of varying size, at Mach numbers from 1·34 to 2·41. Curves are presented for determining the maximum length/radius ratio of a parallel pipe, which will permit supersonic internal flow, in terms of the Mach number of the stream and Reynolds number of the pipe; the curves are given for both laminar and turbulent internal boundary-layers. The effect of inclination of the pipe to the stream is discussed briefly, on the basis of results at one Mach number (1·86).—(1.5.1.1 × 1.1 × 1.2.3.1).

*Cascade flow problems. H. Schlichting. AGARD Report* 93. *February* 1957.

Some of the research work on cascades done in recent years in the Institute of Fluid Mechanics of the Engineering University of Braunschweig are reported. Results of theoretical calculations of loss coefficients of two-dimensional cascades in incompressible flow are included, leading to the characteristic curves of an axial flow compressor, and results are given of pressure distribution measurements on cascades at high subsonic Mach numbers carried out in the new Variable Density High Speed Cascade Wind Tunnel, which allows independent variation of Reynolds number and Mach number.—(1.5.4).

*Flow-turning losses associated with zero-drag external-compression supersonic inlets. R. C. Meyer. N.A.C.A. T.N.* 4096. *October* 1957.

An analysis based on momentum and continuity considerations is used to evaluate the total pressure recovery of zero-wave-drag external-compression inlets for the Mach number range from 1·0 to 4·0.—(1.5.1.4 × 1.2.3.1).

*N.A.C.A.* 65*-series compressor rotor performance with varying annulus-area ratio, solidity, blade angle, and Reynolds number and comparison with cascade results. W. M. Schulze et al. N.A.C.A. T.N.* 4130. *October* 1957.

An axial-flow compressor rotor was tested at low speed and without guide vanes or stators in a 28 in. test compressor. The rotor was designed for a free-vortex type blade loading using N.A.C.A. 65-series 10 per cent-thick aerofoil sections. In these tests, the effects of changes of blade-setting angle, annulus-area ratio, solidity, and Reynolds number on rotor performance were determined for a range of quantity flows. Rotor data are compared with estimations based on uncorrected and corrected cascade data.—(1.5.2.1).

LOADS—*see* WINGS AND AEROFOILS
MISSILES

PERFORMANCE ESTIMATION—*see* MISSILES

STABILITY AND CONTROL—*see also* WINGS AND AEROFOILS

*Low-speed wind-tunnel measurements of longitudinal oscillatory derivatives on three wing plan-forms. G. F. Moss. R. & M. No.* 3009. 1957.

Tests have been made on a 90° apex delta wing, a 60° swept-back wing and a 40° swept-back wing to obtain values of longitudinal oscillatory derivatives for various frequencies.

NOTE.—*The figures in parenthesis at the end of each Summary are for office use only.*

amplitudes and Reynolds numbers. Values of $m_{\dot{\theta}}$ were obtained for all three models and $z_w$ for the delta model using free oscillation methods. In addition, $z_q$, $z_{\dot{\theta}}$, $m_\theta$ and $m_{\dot{\theta}}$ were measured by a forced oscillation method for the delta wing and the 60° swept-back wing.—(1.8.2.2×1.10.2.2).

*An approximate method of estimating the effect of elastic deformability of the aircraft structure on the manoeuvre point. Parts I and II. H. Fingado and A. S. Taylor. R. & M.* 3019. 1957.

Part I deals with the effect on manoeuvre point of wing deformability, including the effects of sweep. Part II deals mainly with the effects of deformability of the fuselage and of an unswept tail unit, but at the same time applies some results of Part I to determine the effect of wing deformability on the fuselage and tailplane contributions to stability.—(1.8×2).

*An investigation on the influence of amplitude, sidewash effects and frequency on the aerodynamic derivatives of a model oscillating continuously in yaw. J. Yff. N.L.L. T.N. F.*204. *July* 1957.

Results of continuous oscillation tests of various model configurations are presented as a function of the yawing amplitude. These tests were made to investigate a hypothesis about the non-linear influence of the wing-body interference on the aerodynamic yawing stability derivatives. To determine quantitatively the side-wash characteristics due to wing-body interference, static measurements have also been performed. All tests were at a speed of 71 m/sec. and $Re = 1{\cdot}6\times10^6$ while the angle of incidence was 5° and the oscillation frequency 2 c.p.s., except for some configurations which have been tested also at 1·3, 1·6, and 2·7 c.p.s.—(1.8.1.2).

*Effect of ground proximity on the aerodynamic characteristics of a four-engine vertical-take-off-and-landing transport-airplane model with tilting wing and propellers. W. A. Newsom. N.A.C.A. T.N.* 4124. *October* 1957.

An investigation was conducted on an aeroplane model with the wing at an angle of incidence of 90° for a range of heights of the model above the ground and included force tests and tuft studies of the flow field caused by the propeller slipstream.—(1.8.2.2).

*Wind-tunnel investigation of the static lateral stability characteristics of wing-fuselage combinations at high subsonic speeds. Taper-ratio series. J. W. Wiggins and P. G. Fournier. N.A.C.A. T.N.* 4174. *October* 1957.

The aerodynamic characteristics in sideslip of wing-fuselage combinations are presented. The wings have a sweep angle of 45° at the quarter-chord line, aspect ratio of 4, taper ratios of 0·3, 0·6, and 1·0, and a N.A.C.A. 65A006 aerofoil section. The tests covered a Mach number range from 0·4 to 0·95, a Reynolds number range from $1{\cdot}7\times10^6$ to $3{\cdot}2\times10^6$, and an angle-of-attack range from −3° to 24°.—(1.8.1.2).

THERMO-AERODYNAMICS—*see also* BOUNDARY LAYER
MISSILES

*Dynamics of a dissociating gas. III—Non-equilibrium theory. N. C. Freeman. AGARD Report* 133. *July* 1957.

The equilibrium theory of an "ideal dissociating" gas discussed by Lighthill (1956) is extended to non-equilibrium thermodynamic conditions by postulating a rate equation for the dissociation process (including the effects of recombination). The behaviour of the gas in flow through strong normal shock waves and past bluff bodies is studied under non-equilibrium conditions.—(1.9×1.2.3).

*A study of the motion and aerodynamic heating of missiles entering the earth's atmosphere at high supersonic speeds. H. J. Allen and A. J. Eggers. N.A.C.A. T.N.* 4047. *October* 1957.

A simplified analysis of the velocity and deceleration history of missiles entering the earth's atmosphere at high supersonic speeds is presented. The results of this motion analysis are employed to indicate means available to the designer for minimising aerodynamic heating. The heating problem considered involves not only the total heat transferred to a missile by convection, but also the maximum average and local time rates of convective heat transfer.—(1.9×25.2).

*Heat transfer and boundary-layer transition on two blunt bodies at Mach number* 3·12. *N. S. Diaconis et al. N.A.C.A. T.N.* 4099. *October* 1957.

Local heat-transfer values were measured on a hemisphere-cone-cylinder and a 120° included-angle cone-cylinder at a Mach number of 3·12.—(1.9.1×1.1.2.4).

WINGS AND AEROFOILS—*see also* STABILITY AND CONTROL
TESTING AND INSTRUMENTS

*Low-speed wind-tunnel tests on two thin cranked wings with 60-deg sweepback inboard. C. J. Mansell. R. & M. No.* 2995. 1957.

The Reynolds number of the tests was about $2{\cdot}3\times10^6$ and the Mach number 0·18.—(1.10.2.2×1.8).

*Initial lift of finite aspect-ratio wings due to a sudden change of incidence. D. E. Lehrian. R. & M.* 3023. 1957.

The initial lift for wings of finite aspect ratio due to a sudden unit change of incidence is considered and, by the use the method developed in *R. and M.* 2117, the ratio of the initial to the final values of the lift is determined for rectangular wings and cropped delta wings of taper ratio 1/7.—(1.10.1.2×1.6.3).

*An investigation of the velocity distribution around the nose of the aerofoil with a flap. I. H. Rettie. R. & M.* 3027. 1957.

The velocity distribution around the nose of a N.A.C.A. 0015-64 aerofoil was found by experiment and that around the nose of a Piercy 15/40 aerofoil by calculations for various angles of incidence and flap deflection. It was established that at all incidences this velocity distribution is a function of the position of the stagnation point, irrespective of the flap deflection. This result is shown to be true generally, and it is suggested that use might be made of it in the design of a lift-coefficient meter, which could also be used to give warning of a stall.—(1.10).

*The effects of taper on the supervelocities on three-dimensional wings at zero incidence. K. W. Newby. R. & M. No.* 3032. 1957.

Relationships have been derived for expressing the velocities on three-dimensional tapered wings at zero incidence in terms of the velocities on untapered infinite swept wings. The theoretical investigation of the effects of taper is confined to simple wings having aerofoil sections formed by cubic or parabolic arcs; some experimental evidence is given to show that the results of this investigation can probably be applied quantitatively to wings having conventional aerofoil sections. A calculation method is outlined for applying the results obtained for wings having parabolic-arc aerofoil sections, to wings having arbitrary section shapes.—(1.10.1.2).

*A subsonic lifting-surface theory for low aspect-ratio wings. C. R. Taylor. R. & M.* 3051. 1957.

A method is developed for determining the loading on low-aspect ratio wings. By allowing for downstream effects on the flow at a station on the wing and for the trailing-edge condition, the method improves on R. T. Jones' theory for wings of very small aspect ratio. Calculations have been made for thirteen different wings and a comparison with other methods of solution is given for some cases.—(1.10.1.2×1.6.1).

*Curves suitable for families of aerofoils with variable maximum thickness position, nose radius, camber and nose droop. L. H. Tanner. C.P. No.* 358. 1957.

It is shown that it is possible to design geometric curves, given by single explicit equations, and having shapes suitable for use as aerofoil sections. A family of sharp-nosed sections with variable maximum thickness position is described and a method of rounding the leading edge of any sharp-nosed section suggested. Finally a family of curves with camber and nose droop is given. The methods used could be adapted to produce other section variations if required.—(1.10).

*Experimental investigation of the pressure distribution at the centre-section of a sweptback wing at high subsonic speeds. T. E. B. Bateman and A. J. Lawrence. C.P. No.* 367. 1957.

The measurements have been made on a 40° swept-back wings over the Mach number range 0·50 to 0·94 at zero incidence.—(1.10.2.2).

*A note on the cause of the nose stall of thin wings. D. G. Hurley and N. Ruglen. A.R.L. Note A.*162. *May* 1957.—(1.10.2.1).

*Elliptic cones alone and with wings at supersonic speeds. L. H. Jorgensen. N.A.C.A. T.N.* 4045. *October* 1957.

The results of an experimental investigation to determine the aerodynamic characteristics of elliptic cones alone and in combination with triangular wings are given. Data are presented for Mach numbers 1·97 and 2·94 and a Reynolds number of $8 \times 10^6$ (based on cone length). Comparisons of theory with experiment are made.—(1.10.2.2 × 1.2.3).

*Experimental investigation of lift, drag, and pitching moment of five annular airfoils. H. S. Fletcher. N.A.C.A. T.N.* 4117. *October* 1957.

Results are presented of tests made to determine the lift, drag and pitching-moment characteristics of a family of annular aerofoils. The five annular aerofoils tested covered aspect ratios of 1/3, 2/3, 1·0, 1·5, and 3·0. Estimated lift-curve slopes and induced drag coefficients are presented and compared with experimental values.—(1.10.2.2 × 1.8.2.2).

### Helicopter Aerodynamics

*An analysis of the longitudinal stability and control of a single-rotor helicopter. F. O'Hara. R. & M.* 2958. 1957.

An analysis is given of the growth of acceleration in a pull-out and assessment of estimated acceleration curves in terms of the National Advisory Committee for Aeronautics "divergence requirement" suggests that the latter may be satisfied if $H_m$ has a small positive value. Further evidence on this point will be obtained in tests now being made on a number of helicopters to study the correlation of stability and control characteristics and pilots' impressions of the handling qualities. Extension of the theory to stick-free longitudinal stability depends on knowledge of the rotor forces on the control plane and the analysis of these forces is being considered.—(1.11.2).

### Testing and Instruments

*Scale effects at high subsonic and transonic speeds, and methods for fixing boundary-layer transition in model experiments. A. B. Haines et al. R. & M. No.* 3012. 1957.

Several methods which can be used to fix transition are described and the results obtained by using them are compared. In general, in experiments in two-dimensional flow, good agreement is found, and explanations can be advanced for cases in which discrepancies occur. Several uncertainties and difficulties that arise in fixing transition are discussed and illustrated by examples. For three-dimensional tests, examples are given of the spurious results that have been avoided successfully by fixing transition, of the conditions where even at low Reynolds numbers artificial fixing of transition may not be necessary to give a turbulent boundary-layer ahead of the shock, and of the conditions under which there are some doubts whether the methods used for fixing transition have been satisfactory.—(1.12.1.2 × 1.10.2).

*On some aspects of the use of shock tubes in aerodynamic research. B. D. Henshall. R. & M. No.* 3044. 1957.—(1.12.1.3).

*Ultrasonic relaxation and attenuation in freons, in relation to their use in supersonic wind tunnels. Y. Miyahara and E. G. Richardson. C.P.*356. 1957.—(1.12.1.3).

*Turbulence measurements in supersonic flow with the hot-wire anemometer. B. Wise and D. L. Schultz. C.P. No.* 366. 1957.

Previous measurements of turbulence under compressible flow conditions have shown that fluctuations in stagnation temperature can occur. Determination of the mass flow fluctuation level is thus rendered more difficult since the hot-wire is sensitive to both variables, and an improved technique is necessary to distinguish between them. The apparatus developed is described and some experimental results presented to illustrate the method used to separate the two variables.—(1.12.6.3).

*Icing measurements with a single rotating cylinder. C. K. Rush and R. L. Wardlaw. N.A.E. Report LR-*206. *September* 1957.

A method is presented for calculating the rate of icing and cloud water content from the ice accretion on a single rotating cylinder. The effect of catch efficiency and blow-off are discussed. Typical charts are presented for use with a particular size of cylinder.—(1.12 × 24).

*The blow-down wind tunnel at Aachen. A. Naumann et al. AGARD Report* 69. *August* 1956.

The construction, installation and operation of the first stage of the blow-down wind tunnel at Aachen is described. The general layout of the complete tunnel, as finally intended, is also described, and a simple analytical estimation is given of the aerodynamic performance of such a tunnel.—(1.12.1.3).

*The hypersonic facility of the Polytechnic Institute of Brooklyn and its application to problems of hypersonic flight. A. Ferri and P. A. Libby. AGARD Report* 136. *July* 1957.

Engineering problems connected with hypersonic flight are reviewed. The utilisation of a hypersonic wind tunnel with stagnation temperatures of 3,000° to 5,000°R for the solution of these problems is discussed. The hypersonic facility of the Polytechnic Institute of Brooklyn is described; the design considerations of the heating system and the demonstrated and expected performance are presented. Finally, some of the testing techniques developed in connection with conventional testing at a Mach number of 6 and with shrouded model testing are reviewed.—(1.12.1.3).

*Research in the U.S. Naval Ordnance Laboratory ballistics ranges. P. A. Thurston. AGARD Report* 137. *July* 1957.

A brief account of the complementarism of wind tunnels and ballistics ranges as tools for aerodynamics research is given. There is a general description of the ballistics ranges in use at the U.S. Naval Ordnance Laboratory and a variety of examples to illustrate the research programme in these ranges.—(1.12.1.3)

*The use of gun-launched models for experimental research at hypersonic speeds. A. Seiff. AGARD Report* 138. *July* 1957.

Two ways of extending gun-launched model tests far into the hypersonic Mach number range have been exploited. One is to use a counter-current supersonic air stream flowing opposite to the free-flying model, by means of which a Mach number of 13 can be attained. The second approach replaces powder gas as a propellant in guns by compression-heated helium. In this way, short, light-weight projectiles have been propelled at speeds in excess of 17,000 ft./sec. In combination, the high performance gun and the counter-current air stream give still higher Mach numbers, of the order of 25.—(1.12.1.3).

*The light gas hypersonic gun tunnel at A.R.D.E., Fort Halstead, Kent. R. N. Cox and D. F. T. Winter. AGARD Report* 139. *July* 1957.

A description is given of an intermittent hypersonic tunnel having a running time of hundreds of milliseconds. The tunnel employs a gun as a compressor. A light piston is driven at supersonic velocity in a closed barrel, and the multiple shock reflections between the piston and the closed end of the barrel provide a high temperature, high pressure stagnation region of gas, which is allowed to expand through a convergent-divergent nozzle when the piston has come to rest. A theoretical study is given of the piston motion and of the shock reflection process.—(1.12.1.3).

*N.A.C.A. hypersonic rocket and high-temperature jet facilities. P. E. Purser and A. C. Bond. AGARD Report* 140. *July* 1957.

The extension of the rocket-model research of the N.A.C.A. to hypersonic speeds is discussed and several high-temperature jet facilities assembled by the N.A.C.A. for research on structures and materials briefly described. Of the jets described, one is essentially the exhaust of a ram-jet engine and provides stagnation temperatures up to 3,000°F, one is a rocket engine and provides stagnation temperatures up to 4,200°F, one is a 4,000°F air jet using a ceramic heat exchanger, and one is an electric-arc-heated jet with stagnation temperatures of over 15,000°F.—(1.12 × 1.12.2).

*Hypersonic shock tube equipment at the National Physical Laboratory. D. L. Schultz and B. D. Henshall. AGARD Report* 147. *July* 1957.

Descriptions are given of an existing shock tube fitted with a flow expansion nozzle giving a Mach number of approximately 9, and of a proposed larger tube. Performance charts

illustrate the conditions for which full-scale flight might be simulated. A unified electronic timing and recording system is also described.—(1.12.1.3).

*External interference effects of flow through static-pressure orifices of a N.A.C.A. airspeed head at a Mach number of 3. N. S. Silsby. N.A.C.A. T.N.* 4122. *October* 1957.

Wind tunnel tests have been made to determine the static-pressure error resulting from external interference effects of flow through the static-pressure orifices of an N.A.C.A. air speed head at a Mach number of 3 at 0° angle of attack. Results were obtained for flow both into and out of the orifices for a range of mass flow from 0 to about $4{\cdot}2\times10^{-6}$ slugs per second.—(1.12.5).

*Development of a piston-compressor type light-gas gun for the launching of free-flight models at high velocity. A. C. Charters et al. N.A.C.A. T.N.* 4143. *November* 1957.

The design of a light-gas gun developed at the Ames Aeronautical Laboratory to launch small models for aerodynamic tests is given and the analysis of its performance. The results of the initial firing trials are discussed. A velocity of 15,400 ft./sec. was reached.—(1.12.1.3).

## AEROELASTICITY

*See also* CONTROL SURFACES
STABILITY AND CONTROL

*Aerodynamic flutter coefficients for subsonic, sonic and supersonic flow (linear two-dimensional theory). P. F. Jordan. R. & M. No.* 2932. 1957.

The four main aerodynamic force coefficients for an oscillating two-dimensional wing are presented in the form of tables and isometric graphs over the ranges 0 to 2 of Mach number and 0 to 1·4 of frequency parameter based on the wing chord; the present sonic solution and existing subsonic and supersonic solutions have been supplemented by interpolated values for Mach numbers between 0·7 and unity.—(2).

*Derivative measurements and flutter tests on a rectangular wing with a full-span control surface, oscillating in modes of wing roll and aileron rotation. W. G. Molyneux and F. Ruddlesden. R. & M. No.* 3010. 1957.

Details are given of tests to measure the aerodynamic coefficients for a rectangular wing with a full-span aileron oscillating in modes of wing roll and aileron rotation. A new technique was used in which aileron rotation was geared to wing roll so that oscillation occurred in both degrees of freedom simultaneously. The measured coefficients are compared with those derived from two-dimensional theory, and with coefficients estimated by an empirical method. The agreement with theory is poor but the estimated coefficients agree well with those measured. Flutter calculations for the system were made, using both measured and theoretical derivatives, and the results are compared with flutter test results. The calculated flutter speed using measured derivatives agrees closely with that measured, whereas using theoretical derivatives the agreement is poor.—(2).

*The aerodynamic effects of aspect ratio and sweepback on wing flutter. W. G. Molyneux and H. Hall. R. & M. No.* 3011. 1957.

Tests to obtain direct measurements of the aerodynamic effects of aspect ratio and sweepback on wing flutter are described. The tests were made on rigid wings with root flexibilities.—(2).

## AIRCRAFT OPERATION

*Analysis of operational airline data to show the effects of airborne weather radar on the gust loads and operating practices of twin-engine short-haul-transport airplanes. M. R. Copp and W. G. Walker. N.A.C.A. T.N.* 4129. *November* 1957.

Air speed, altitude, and acceleration data were obtained from transports flown by one airline before and after the installation of airborne weather radar. A comparison of the results indicated that the magnitudes of the largest gust velocities and gust accelerations experienced for a given number of flight miles during radar operations were approximately 25 per cent less than those experienced before the radar equipment was installed.—(5×24).

## FLIGHT TESTING

*Flight tests on a Vampire Mk. 5 with a redesigned anti-snaking rudder. W. J. G. Pinsker. R. & M.* 3014. 1957.—(13.2).

## FUELS AND LUBRICANTS

*See* THERMODYNAMICS

## HYDRODYNAMICS

*Investigation of high length/beam ratio seaplane hulls with high beam loadings. Hydrodynamic Stability Part* 20. *The effect of slipstream on stability and spray characteristics. D. M. Ridland. C.P. No.* 349. 1957.

The effects of slipstream on longitudinal stability, spray and elevator effectiveness are deduced from tests on four configurations of the basic model of the series which differed only in aerodynamic details. It was found that in general increases in thrust coefficient improved longitudinal stability. spray characteristics and elevator effectiveness. A method of applying the detailed results to the stability limits of other models of the series is outlined.—(17.2).

*Assessment of the possibility of using suction to inhibit cavitation on cylindrical sections. G. E. Thomas. C.P. No.* 362. 1957.

The possibility of using suction to reduce the suction peak in the pressure distribution on a body moving in a fluid is assessed, with a view to delaying the onset of cavitation. Two cases are considered: (*i*) flow past elliptic cylinders with suction applied over the entire forward half, and (*ii*) flow past circular cylinders with suction applied over limited areas. Results are given of calculations made to determine the effect of varying amounts of suction as well as, in (*ii*), the added effect of changing the location and extent of the suction area.—(17.1).

*Rough-water impact-load investigation of a chine-immersed V-bottom model having a dead-rise angle of* 10°. *M. F. Markey and T. D. Carpini. N.A.C.A. T.N.* 4123. *October* 1957.

A hydrodynamic rough-water impact loads investigation of a fixed-trim V-bottom float is described. Time histories were obtained showing the position of the model relative to the wave throughout the impact and typical examples are presented. The load coefficient was found to vary primarily with the slope of the impacting wave.—(17.2×33.1).

## MATERIALS

*See also* POWER PLANTS
THERMODYNAMICS

*Generalized master curves for creep and rupture. G. J. Heimerl and A. J. McEvily. N.A.C.A. T.N.* 4112. *October* 1957.

The similarity of Larson-Miller master curves for creep rate, rupture life, and creep strain is shown for a number of materials. By the use of the approximate invariance of the product of creep rate and rupture life, the individual master curves are generalised into a single curve which is the same as the master curve for rupture. From this master curve, the creep rate and various amounts of creep strain can be predicted.—(21.1).

*Further observations on the ductility of chromium. H. L. Wain et al. A.R.L. Report MET.* 21. *May* 1957.—(21.2.2).

*The mechanical and engineering properties of commercially available titanium alloys. H. V. Kinsey. AGARD Report* 100. *April* 1957.

The titanium alloys being produced on a commercial basis up to early 1957 are listed with their nominal chemical compositions and manufacturers' designations. An attempt has been made to collect all the published data on the physical, mechanical, and engineering properties of these alloys and to indicate those properties required by the aeronautical design engineer that are not available.—(21.2.2).

*Titanium fabrication. L. P. Spalding. AGARD Report* 96. *April* 1957.

A description is given of the problems encountered and the techniques evolved for their solution during the manufacture

of some 2,000 aeroplanes incorporating about a million pounds of titanium in their designs. Primary emphasis is given to titanium alloy sheet applications.—(21.2.2).

*A method for taper sectioning aluminium and its alloys. N. Ryan. A.R.L. Note MET 5. December* 1956.

Methods of treating and mounting aluminium specimens for taper sectioning were investigated. The most successful method consisted of mounting in a moulding mixture of bakelite and aluminium oxide. Examples of edge profiles produced by this method are given.—(21.6.1).

## MATHEMATICS

*A note on some integrals in aerodynamics. D. E. Williams. C.P.*364. 1957.

Some double integrals in aerodynamics are difficult to evaluate because of the singularities in the range of intergration. The Dirac delta function has been found useful in evaluating such integrals. Some examples of its uses are given.—(22.1×1).

## MECHANICAL ENGINEERING

*An analysis of the effect of several parameters on the stability of an air-lubricated hydrostatic thrust bearing. W. H. Roudebush. N.A.C.A. T.N.* 4095. *October* 1957.

Equations are developed for the motions of a gas-lubricated hydrostatic thrust bearing, and solutions are obtained on a digital computer for air as the gas. Systematic investigations are made of various parameters to determine their effect on bearing stability. Bearing pad volume and rigidity appear as prime controlling factors.—(23.1).

*Effect of lubricant viscosity on rolling-contact fatigue life. T. L. Carter. N.A.C.A. T.N.* 4101. *October* 1957.

The rolling-contact fatigue spin rig was used to obtain life data for groups of steel balls lubricated with a series of paraffin-based mineral oils of varying viscosity. AISI M-1 balls were tested at 100°F and a calculated maximum compressive Hertz stress of 725,000 p.s.i. A continuous trend towards longer life with higher lubricant viscosity was observed.—(23.1×31.2.3.1.8).

## METEOROLOGY

*See* TESTING AND INSTRUMENTS
AIRCRAFT OPERATION

## MISSILES

*See also* THERMO-AERODYNAMICS

*A comparative analysis of the performance of long-range hypervelocity vehicles. A. J. Eggers et al. N.A.C.A. T.N.* 4046. *October* 1957.

A simplified analysis is made of the motion and aerodynamic heating of long-range ballistic-, skip-, and glide-type vehicles. The ballistic vehicle appears relatively attractive because convective heat transfer can be reduced by using blunt shapes. The glide vehicle appears attractive because it has a relatively efficient trajectory, and the possibility of substantial radiative cooling. These vehicles compare favourably, in the sense of the Breguet range equation, to the supersonic aeroplane for very long-range flight.—(25.3×1.9×1.7).

*Motion of a ballistic missile angularly misaligned with the flight path upon entering the atmosphere and its effect upon aerodynamic heating, aerodynamic loads, and miss distance. H. J. Allen. N.A.C.A. T.N.* 4048. *October* 1957.

An analysis is given of the oscillating motion of a ballistic missile which upon entering the atmosphere is angularly misaligned with respect to the flight path. The history of the motion for some example missiles is discussed from the point of the effect of the motion on the aerodynamic heating and loading. The miss distance at the target due to misalignment and to small accidental trim angles is treated. The stability problem is also discussed where the missile is tumbling before atmospheric entry.—(25.2×1.9×1.6.3).

## POWER PLANTS

*Investigations on an experimental air-cooled turbine. Part I—General description of turbine and experimental technique. D. G. Ainley and N. E. Waldren. Part II—Cooling characteristics of blades having a multiplicity of small diameter coolant passages. D. G. Ainley et al. R. & M. No.* 2975. 1957.

Part I gives a general description of an experimental single-stage air-cooled turbine which has been erected and operated at the N.G.T.E. Part II continues with a description of some internally air-cooled nozzle and rotor blades together with an account of some experimental investigations concerning their performance.—(27.1).

*Drop-size distribution for crosscurrent breakup of liquid jets in airstreams. R. D. Ingebo and H. H. Foster. N.A.C.A. T.N.* 4087. *October* 1957.

Photographic and sampling techniques were combined to obtain drop-size data for ranges of injector, liquid, and airstream variables.—(27.4).

*Investigation of some mechanical properties of thermenol compressor blades. D. F. Johnson. N.A.C.A. T.N.* 4097. *October* 1957.

A series of tests were made comparing the mechanical properties of similar compressor blades of AISI type 403 stainless steel and thermenol. Eighth-stage J47 and J65 compressor blades of each material were tested. Modulus of elasticity, modulus of rigidity, and damping were slightly lower for thermenol but thermenol showed better resistance to corrosion by sea water, and the thermenol blades proved equal or superior to the stainless-steel blades in fatigue strength.—(27.1×21.2.1).

*Experimental investigation of cermet turbine blades in an axial-flow turbojet engine. W. C. Morgan and G. C. Deutsch. N.A.C.A. T.N.* 4030. *October* 1957.

In 16 tests to evaluate cermet blades in an axial-flow engine, a maximum blade life of 150 hours at maximum conditions of operation was obtained. Chipping in the tip region was a common characteristic in blade failures. Analysis indicates two probable causes of failure, impact and fatigue. Elimination of excessive carbon in the combustion chambers and avoiding conditions of extreme vibration made an improvement in test blade life.—(27.1×21.3).

*Propellant vaporization as a criterion for rocket-engine design: calculations using various log-probability distributions of heptane drops. R. J. Priem. N.A.C.A. T.N.* 4098. *October* 1957.

Vaporisation rates of *n*-heptane sprays in a rocket combustion chamber were computed for various log-probability distributions and mass-median drop sizes of the spray and for an engine operating with various injection velocities, final gas velocities, initial fuel temperatures, and chamber pressures. Incomplete vaporisation of a small number of large drops may be responsible for the loss in engine performance. Experimental engine performance agrees with the calculations for a spray having a geometric standard deviation of 2·5 and mass-median drop radii of 70 to 280 microns, depending on the type of injector. The calculations of the per cent of fuel vaporised are correlated with an effective chamber length.—(27.3×34.3).

## FATIGUE

*See also* MECHANICAL ENGINEERING
STRUCTURES—LOADS

*The behaviour of fractured copper fatigue specimens on annealing. D. S. Kemsley. A.R.L. Report Met.* 22. *July* 1957.

Fractured rotating-cantilever specimens have been annealed at constant temperature and also at constant heating rate. Hardness, metallographic and X-ray diffraction observations have been made at stages during annealing. In low-stress specimens, deformation markings disappear over a wide temperature range concurrently with a drop in hardness; recrystallisation does not occur. High-stress specimens, in which etching does not reveal striations, recrystallise over a narrow range of temperature, with an associated rapid drop in hardness. This behaviour is comparable to that of a tension specimen similarly annealed.—(31.2.2.5.1.3).

## SCIENCE—GENERAL

*Investigation of a wide angle diffuser with air augmentation for use as a jet muffler. A. J. Campbell. U.T.I.A. T.N.* 15. *August* 1957.

A wide angle diffuser for reduction of the noise of a jet, by taking advantage of the $AV^8$ law was combined with a short

length augmentation system, which, on a full scale muffler, would provide cooling air flow to the diffuser. A theory is developed which predicts qualitatively the induced air flow when a short mixing length is used.—(32.2.3 × 1.5.1).

## STRUCTURES

LOADS—*see also* HYDRODYNAMICS

*A contact extensometer for the recording of flight-load spectra. O. Svenson. AGARD Report* 121. *May* 1957.

A contact extensometer, developed at Darmstadt, Germany, for recording flight-load spectra is described. The varying stress cycles are registered on a film which is run at low speed, and the subsequent analysis can be made by any method desired. The instrument is to be installed in aircraft operated by the German Lufthansa.—(33.1 × 33.3.1).

*Fatigue loadings in flight-loads in the tailplane of a Comet I. A. Burns. C.P. No.* 363. 1957.

Data are presented on the number of load cycles of various magnitudes occurring in the tailplane of a Comet IA during normal ground and flight conditions. The conditions include flight in turbulence, take-off, landing, taxying and ground running of the engine. The relative importance of the loads in the different conditions is illustrated by reference to the loads in a typical flight.—(33.1.2 × 31.3.1).

*The effect of airplane structural flexibility on landing gear loads. J. P. Bentham. NLL-TN S.*478. *December* 1956.

A method is developed to solve the equations of the landing gear and those of the elastic system of the remainder of the aircraft simultaneously. The landing gear strut has a velocity-square damping device and pre-loading by air pressure. Spin-up and spring-back loads are taken into account. The parameters varied in numerical studies are the vertical velocity of descent, number of modes of vibration taken into account and stiffnesses of wing. Two locations of landing gear are investigated. A simple method to estimate the importance of wing flexibility is given.—(33.1).

THEORY AND ANALYSIS

*The matrix force method of structural analysis and some new applications. J. H. Argyris and S. Kelsey. R. & M. No.* 3034. 1957.

The basic principles of the matrix force method of structural analysis are summarised and some new applications of the general theory presented, the special method of cut-outs is established and illustrated and a method for the determination of the stresses in a structure is presented, some of whose components have been modified subsequent to an initial stress analysis.—(33.2.4).

*Tests on eccentrically loaded riveted joints. A. J. Francis and G. L. Belcher. A.R.L. Report SM* 249. *April* 1957.

Tests to failure are described on 15 riveted joints. The load was statically applied with varying initial eccentricities. The strengths obtained were, on the average, 55 per cent in excess of the strengths calculated by the usual procedure for eccentrically loaded splices, but only 3·6 per cent below the strengths predicted by a proposed theory which makes allowance for the development of plasticity in plates and rivets.—(33.2.4.13.1).

*Elastic stability of rectangular sandwich plates under bi-axial compression. J. Guest and J. Solvey. A.R.L. Report SM* 251. *May* 1957.

Buckling loads in the elastic range of rectangular sandwich panels having isotropic cores and faces, with four clamped edges and with two opposite edges clamped and two opposite edges simply-supported, subjected to bi-axial compression, have been computed using Rayleigh's method. Results are tabulated and plotted in terms of non-dimensional parameters. Interaction curves have also been plotted to facilitate practical application.—(33.2.4.6.6).

*Handbook of structural stability. Part I—Buckling of flat plates. G. Gerard and H. Becker. N.A.C.A. T.N.* 3781. *July* 1957.

The various factors governing the buckling of flat plates are critically reviewed and the results summarised in a comprehensive series of charts and tables. Numerical values are presented for buckling coefficients of flat plates with various boundary conditions and applied loadings. The effects of plasticity are incorporated in non-dimensional buckling charts utilising the three-parameter description of stress-strain curves.—(33.2.4.5.6).

*Handbook of structural stability. Part II—Buckling of composite elements. H. Becker. N.A.C.A. T.N.* 3782. *July* 1957.

The local buckling of stiffener sections and the buckling of plates with sturdy stiffeners are reviewed, and the results summarised in charts and tables. Numerical values of buckling coefficients are presented for longitudinally compressed stiffener sections of various shapes, for stiffened plates loaded in longitudinal compression and in shear, and for stiffened cylinders loaded in torsion. Although the data consist primarily of elastic-buckling coefficients, the effects of plasticity are discussed for a few special cases.—(33.2.4.2.6 × 33.2.4.6.6 × 33.2.4.9.6).

*Handbook of structural stability. Part III—Buckling of curved plates and shells. G. Gerard and H. Becker. N.A.C.A. T.N.* 3783. *August* 1957.

Available theories and test data on buckling of curved plates and shells are reviewed. The test data for torsion and external-pressure loadings are correlated in terms of linear buckling theories for both the elastic and inelastic ranges. The cases which exhibit a marked disagreement between linear theory and test data have been analysed by a unified semi-empirical approach which is satisfactory for analysis and design purposes.—(33.2.4.3.6 × 33.2.4.8.6 × 33.2.4.11.6).

*Handbook of structural stability. Part IV—Failure of plates and composite elements. G. Gerard. N.A.C.A. T.N.* 3784. *August* 1957.

Available theories on failure of flat plates are reviewed, and test data on the postbuckling behaviour, effective width, and failure of flat and curved plates are correlated. Test data on the crippling strength of various formed and extruded shapes are reviewed, from which a generalised method of crippling analysis is formulated. This analysis is then applied to a variety of sections and materials in common use.—(33.2.4.0.1).

*Handbook of structural stability. Part V—Compressive strength of flat stiffened panels. G. Gerard. N.A.C.A. T.N.* 3785. *August* 1957.

A generalised crippling analysis for short monolithic panels with formed or extruded stiffeners is presented. Methods are given for determining if riveted panels act in a monolithic manner and for determining the strength of those which do not. The failure modes of intermediate-length and long stiffened panels are discussed and methods given for estimating column strength. Theory and test data on optimum stiffened panels are presented and the various factors of importance in box types of construction are considered.—(33.2.4.6.1).

*Weight-strength studies of structures representative of fuselage construction. J. P. Peterson. N.A.C.A. T.N.* 4114. *October* 1957.

Weight-strength plots, suitable for use in strength and weight analysis in the preliminary design of aircraft construction, are presented for cylindrical shells in bending. Of the types of stiffening considered, sandwich-type stiffening is found to be the most effective, and no decided advantage is found between longitudinal and waffle-like stiffening for minimum-weight designs within the range of structural configurations considered. The ratio of compressive-yield stress to density of material is found to be the most important material parameter for sandwich-type shells, while the density of the material is found to be equally as important for shells with longitudinal or waffle-like stiffening.—(33.2.3.3).

TESTING—*see also* LOADS

*Impact tests of flexible non-metallic aircraft fuel tanks installed in two categories of simulated wing structures. C. V. Bennett and R. J. Schroers. CAA TDR No.* 291. *January* 1957.

Tests have been conducted to correlate the ability of a non-metallic aircraft fuel tank to resist rupture under impact loads with material strength and/or energy-absorbing properties. The non-metallic tanks were housed in two types of simulated wing structures with chordwise strength characteristics equivalent to those of a modern twin-engined and a modern four-engined aeroplane.—(33.3.1).

WEIGHT ANALYSIS AND CONTROL

*Merit indices for structural materials. H. B. Howard. AGARD Report* 105. *April* 1957.

Merit indices are derived to give weight efficiency for a number of simple structural properties. These include direct strength and stiffness, the buckling of struts, the wrinkling of tubes, thermal stress and deformation. The indices are applied to typical aluminium and titanium alloys and to a stainless steel at temperatures up to 400°C. A simplified form of multi-web wing box is studied, the analysis extending to both stress and buckling failure. An optimum curve for any material is derived. The same three materials are compared for the same temperature range. The comparison of weight for a given torsional rigidity is made by reference to a tube of elliptic cross-section for which simple generalised expressions are found.—(33.4.1).

## THERMODYNAMICS

*See also* POWER PLANTS

*Internal air-cooling for turbine blades. A general design survey. D. G. Ainley. R. & M.* 3013. 1957.

The degree of cooling that might be achieved in gas-turbine blades with simple internal air-cooling is surveyed with a view to pin-pointing the essential requirements for effective cooling with small quantities of cooling air. A shape parameter $Z$ (defined as $(S_c/c)^{1.2}/(A_c/c^2)$) is derived which forms a useful figure of merit for comparing the relative efficiences of various cooling passage configurations. Potentialities of blades with laminar-cooling and turbulent-cooling flow in the cooling passages are compared.—(34.3.4 × 1.5.3).

*Thermodynamic properties of air: tables and graphs derived from the Beattie-Bridgeman equation of state assuming variable specific heats. R. E. Randall. ASTIA AD*-135331. *August* 1957.—(34.1).

*Thermodynamic properties of gases: equations derived from the Beattie-Bridgeman equation of state assuming variable specific heats. R. E. Randall. ASTIA AD*-135332. *August* 1957.

The Beattie-Bridgeman equation of state was used to develop the equations of several of the thermodynamic properties and flow process correction factors for gases. The increase in the specific heats due to the vibration of diatomic molecules was included by assuming the molecules to be perfect harmonic oscillators. Thermodynamic and flow process equations are theoreticaly developed. The particular flow processes investigated were isentropic expansion and flow through normal shock waves.—(34.1).

*Propellant vaporization as a criterion for rocket engine design; calculations of chamber length to vaporize a single n-heptane drop. R. J. Priem. N.A.C.A. T.N.* 3985. *July* 1957.

Calculations were made for *n*-heptane drops using various drop sizes, injection velocities, final gas velocities, initial fuel temperatures, and chamber pressures to show how these variables would affect the vaporisation rate and the chamber lengths required to vaporise the drops.—(34 × 14).

*Experimental and calculated histories of vaporizing fuel drops. R. J. Priem et al. N.A.C.A. T.N.* 3988. *August* 1957.

The results of an experimental and theoretical investigation of the vaporisation of fuel droplets in heated air under atmospheric pressure are presented.—(34 × 14).

*Effects of thermal relaxation and specific-heat changes on measurements with pneumatic-probe pyrometer. P. W. Kuhns. N.A.C.A. T.N.* 4026. *July* 1957.

Equations are derived which evaluate the effects of thermal vibrational relaxation and specific heat changes on pneumatic-probe pyrometer measurements in the region 1,000° to 3,000°K and 0·3 < Mach number < 2·0. Examples are given for typical probes in two combustion mixtures.—(34.2.4).

*Preliminary investigation of propane combustion in a 3-inch-diameter duct at inlet-air temperatures of* 1400° *to* 1600°F. *E. A. Lezberg. N.A.C.A. T.N.* 4028. *July* 1957.

Ignition delays and combustion efficiences were determined for propane injected into a heated air stream.—(34.1).

*Stability limits and burning velocities for some laminar and turbulent propane and hydrogen flames at reduced pressure. B. Fine. N.A.C.A. T.N.* 4031. *August* 1957.

The stability behaviour for propane-oxygen-nitrogen, hydrogen-argon-"air", and hydrogen-helium-"air" burner flames is similar to that previously found for hydrogen-air flames. The propane-oxygen-nitrogen system shows greater reactivity based on flashback than does the hydrogen-air system, but less stability toward reduced pressures. The pressure exponent of burning velocity was nearly the same as for hydrogen-air flames. Critical boundary velocity gradients for laminar and turbulent flashback and turbulent blow-off were correlated with pressure.—(34.1).

*A method of computing the transient temperature of thick walls from arbitrary variation of adiabatic-wall temperature and heat-transfer coefficient. P. R. Hill. N.A.C.A. T.N.* 4105. *October* 1957.

Formulas to determine the transient temperature of thick walls with an arbitrary time variation of adiabatic-wall temperature and heat-transfer coefficient are presented. The special cases of the thin wall and the infinitely thick wall are also discussed. A simple formula for determining the heat flow from a known temperature history is also given. Heat transfer in a direction parallel to the surface is assumed to be negligible and constant wall properties are assumed.—(34.3).

*Theory and apparatus for measurement of emissivity for radiative cooling of hypersonic aircraft with data for Inconel and Inconel X. W. J. O'Sullivan and W. R. Wade. N.A.C.A. T.N.* 4121. *October* 1957.

The importance of radiation as a means of cooling high speed aircraft is discussed to show the need for measurements of the total hemispherical emissivity of surfaces. The theory underlying the measurement of the total hemispherical emissivity of surfaces is presented, readily duplicable apparatus for performing the measurements is described, and measurements for stably oxidised Inconel and Inconel X are given for the temperature range from 600°F to 2000°F.—(34.3.1 × 21.2).

*Effect of initial mixture-temperature on burning velocity of hydrogen-air mixtures with preheating and simulated preburning. S. Heimel. N.A.C.A. T.N.* 4156. *October* 1957.

Laminar burning velocities were determined from schlieren photographs of open flames. The temperature was raised (1) by preheating of the hydrogen-air mixtures and (2) by simulated adiabatic preburning of part of the hydrogen in air at 300°K so that initial temperatures of 600° and 700°K would be attained. The temperature dependence of burning velocity was determined for both preheated and preburned mixtures.—(34.1.1).

*Study of some burner cross-section changes that increase space-heating rates. D. R. Boldman and P. L. Blackshear N.A.C.A. T.N.* 4162. *November* 1957.

A two-dimensional glass-walled combustor fed with a homogeneous combustible mixture was used to study the effect of area blockage on heat release. The blockage at a single V-gutter flame holder was varied from 12·5 to 75 per cent with negligible effect on combustion efficiency. When a small flame holder was used and a 62·5 per cent restriction introduced down-stream, the heat-release rate underwent a three- to four-fold increase. Some effects of downstream blockage shapes are given.—(34.1.1).

# APPOINTMENTS

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